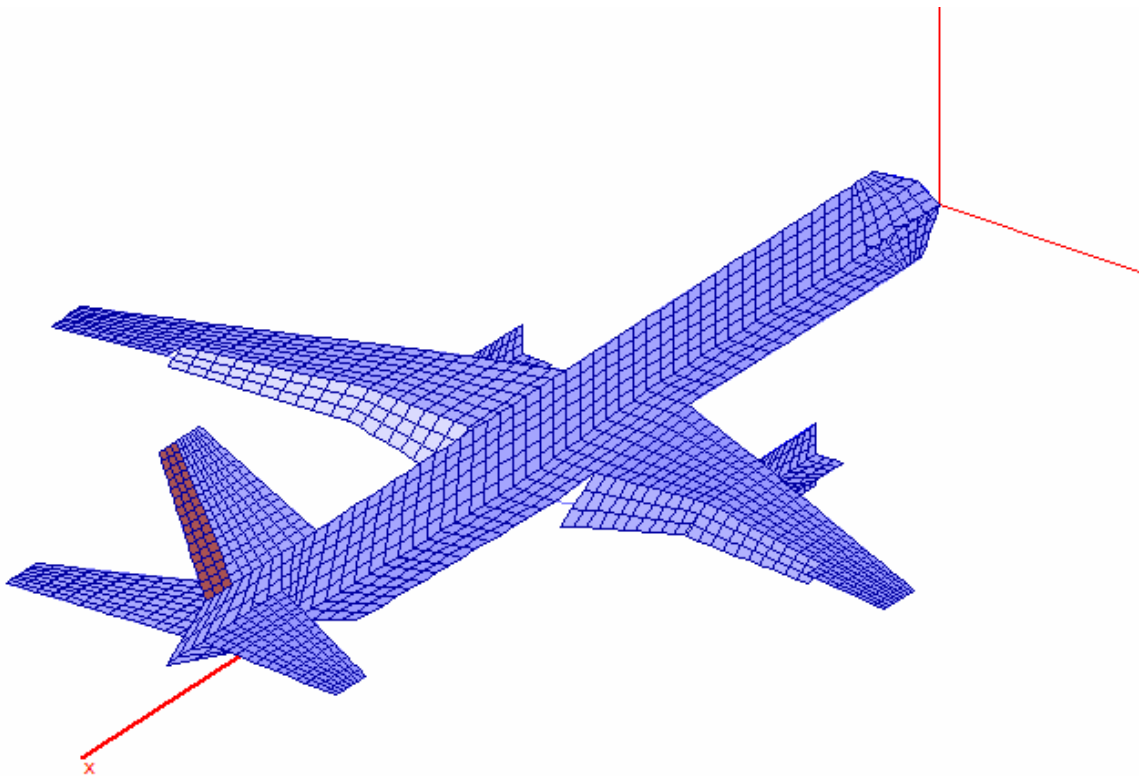


**SURFACES**

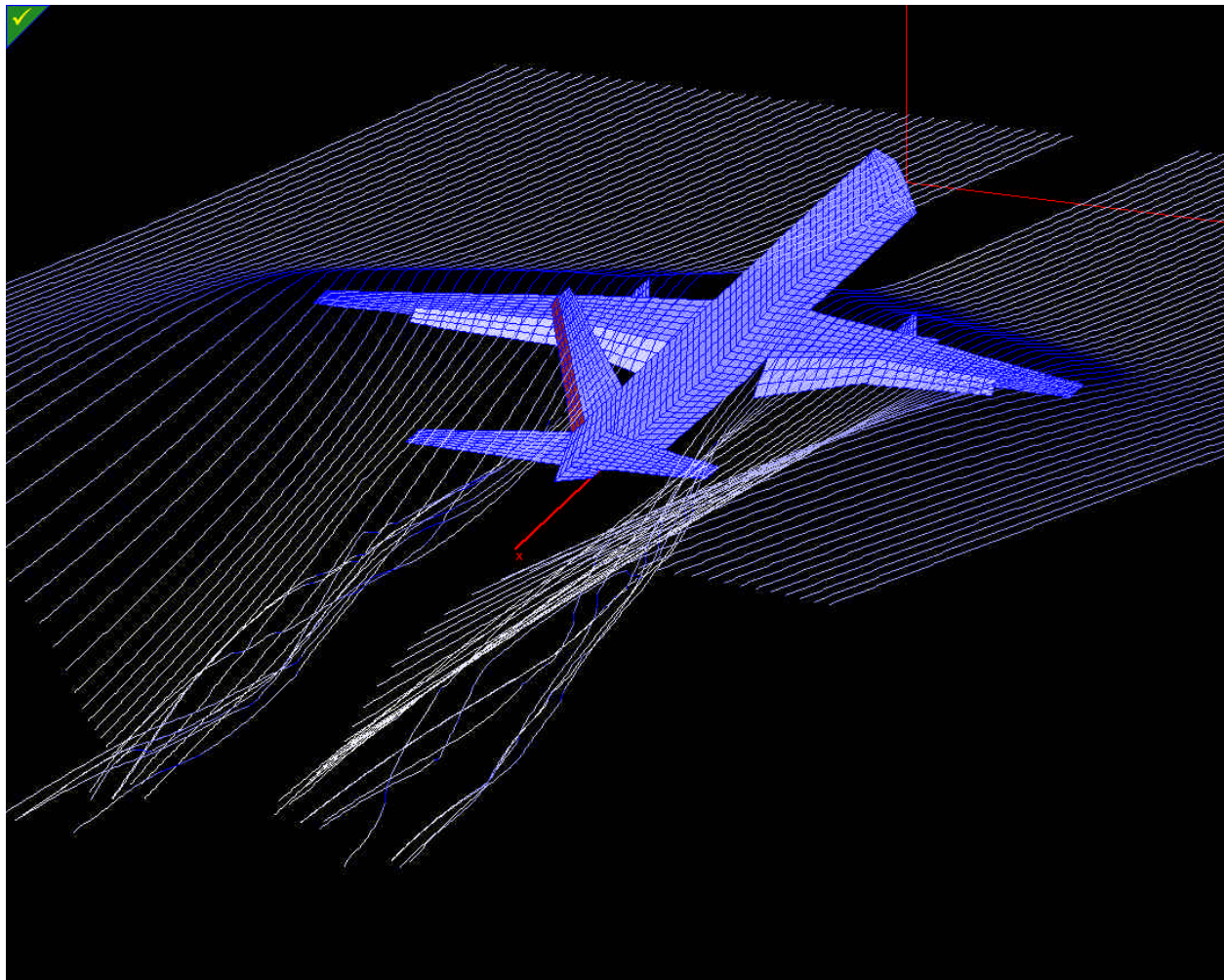
**VORTEX LATTICE MODULE**



**User Manual**

# SURFACES

## Vortex-Lattice Module



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## SURFACES – Vortex-Lattice Module

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# INTRODUCTION

Thank you for purchasing **SURFACES**.

We are certain you will find **SURFACES** priceless for your aircraft design projects. **SURFACES** was developed in real aircraft design environment and is loaded with highly developed tools that give you answers quickly. We consider the program analogous to an extremely sophisticated airplane calculator. Create a model of your aircraft and then use **SURFACES** to extract hard-to-get information about it. Stability derivatives, loads, performance parameters are just the beginning of your discoveries. You can extract in a matter of seconds some super complicated parameters that would take a trained aerospace engineer weeks to calculate using classical methods. Use the extra time to study variations of your design to make it even better for its intended mission. Whatever the design task, **SURFACES** will save you weeks if not months of work.

**SURFACES** is the ultimate tool for anyone designing subsonic aircraft, whether it be a professional aerospace engineer or the designer of homebuilt aircraft. **SURFACES** is not just user friendly, it provides you with very powerful features to help design your aircraft.

**SURFACES** uses a Three-Dimensional Vortex Lattice Method (VLM) to solve the airflow around an aircraft and extract an incredible amount of information from the solution. Plot the flow solution to better understand how the flow behaves around the airplane.

**SURFACES** is the perfect solution in any preliminary design environment, or to reverse engineer existing airplanes. It allows you to quickly extract loads and stability and control data.

**SURFACES** allows you to swiftly model any aircraft. Do you have a three-view drawing of your favorite aircraft? Simply import it in to the environment and scale it up. No pencils, rulers, or calculators are needed for scaling up the model. You do it all from within **SURFACES**. It's as easy as clicking a mouse button.

**SURFACES** determines most stability derivatives and, when used with the built-in Aircraft Datasheet feature, allows you to perform very sophisticated dynamic stability analyses. Import stability derivatives directly from your Vortex-Lattice analyses into an Aircraft Datasheet and plot the aircraft's Short Period, Phugoid, Spiral Stability, Rolling Convergence, and Dutch Roll modes. You can even simulate the dynamic response of the aircraft in real time!

**SURFACES** allows you to incorporate all the details of your design, such as airfoil properties, wing twist, dihedral, multiple lifting surfaces, asymmetric geometries, winglets, deflection of control surfaces and high lift devices. **SURFACES** even allows you to account for engine forces as functions of angle-of-attack, airspeed and altitude, whose properties are taken into account when determining trim or stability derivatives.

**SURFACES** allows you to extract surface pressures, forces and moments, force and moment coefficients, distributed loads, section lift coefficients, and create shear, moment and torsion diagrams on the model.

**SURFACES** comes with video tutorials. You will be working on your own airplane in 30 minutes or less.

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## "Vortex Lattice Methods" Why Should You Care?

By Mike Garton

Some of the latest glider designs are advertised as having computer optimized wings. For instance ads for the Sapphire, Psyko, Laser, and Edge all list it as a design feature. NSP's ad mentions the "LinAir" program, which uses a form of computational fluid dynamics that we aerospace engineers call "vortex lattice methods" or "simple panel codes". There is not space here to discuss how these codes work (and perhaps not interest either) but I will briefly describe what can be done with these programs and what it means for the pilot. If I lose you in technical jargon, just skip to the last two paragraphs.

A vortex lattice program takes a wing planform, wing-twist, and angle of attack as inputs. Using this information it calculates the induced velocity field surrounding the wing including the effect of tip vortices. It is somewhat non-intuitive, but the angle of attack of a wing is not simply the angle between your root chord and your tailboom. The wing "induces" some vertical components of velocity that change the "effective" angle of attack. Generally the induced angle is smaller at the root of the wing and larger at the wing tips. A tip vortex will add a downward component to the air above the wing tip. This causes the "effective angle of attack" of most wing tips to be reduced. This is one form of aerodynamic wash out. A vortex lattice program allows a designer to quantify these effects, before the plane is built and without the need for a wind tunnel. The use of this tool does not guarantee a good wing. Like any tool, it still takes wisdom and proper application to get good results. This particular tool is usually reserved for graduate degree aerospace engineers with specialization in computational fluid dynamics.

Some of the things a glider designer can do with this program are to: 1. Minimize induced drag (drag due to tip vortices), 2. Manage which part of the wing will stall first, 3. Given a planform, refine its twist distribution, and 4. Calculate the local flow direction on the stab including downwash from the wing. In general, the refined wings have nearly elliptical chord distributions with finite tip chords (no big surprise here). Aerospace Engineers will assert that elliptical lift distributions DO result in the minimum possible induced drag for low speed wings. At our low Reynold's numbers, a truly elliptical chord distribution does NOT result in an elliptical lift distribution. At low speeds on a truly elliptical winged model, the air flow will separate near the wing tip, leading to too little lift in that region and tip stall. This is why the refined sailplanes tend to have finite tip chords. The nearly elliptical wing has another beneficial quality. The downwash angle is relatively constant along the span. This means the whole wing is flying at the same "effective" angle of attack. A constant angle of attack is good because no part of the wing will stall early and the wing can achieve a high average lift coefficient. When any section of the wing stalls, it will usually propagate sideways and stall the entire wing. As an example, a straight taper wing with its uneven effective angle of attack will stall at an average lift coefficient roughly 20% lower than the computer refined four taper wing. I am assuming that the designer of the four taper wing used the vortex lattice code properly.

So what might a pilot notice in flight when flying one of these planes refined with a vortex lattice code? Most pilots won't notice the differences. After trimming the plane, an expert pilot should notice that the launch is steeper because the wing can pull a higher lift coefficient before stalling. The sink rate and glide ratio should be a tweak better as well. We are only talking a couple percent decrease in drag over the "eye balled" planforms, but every little bit helps. The plane should be able to fly slower than other planes with the same airfoil and wing loading, again because of the higher available lift coefficient.

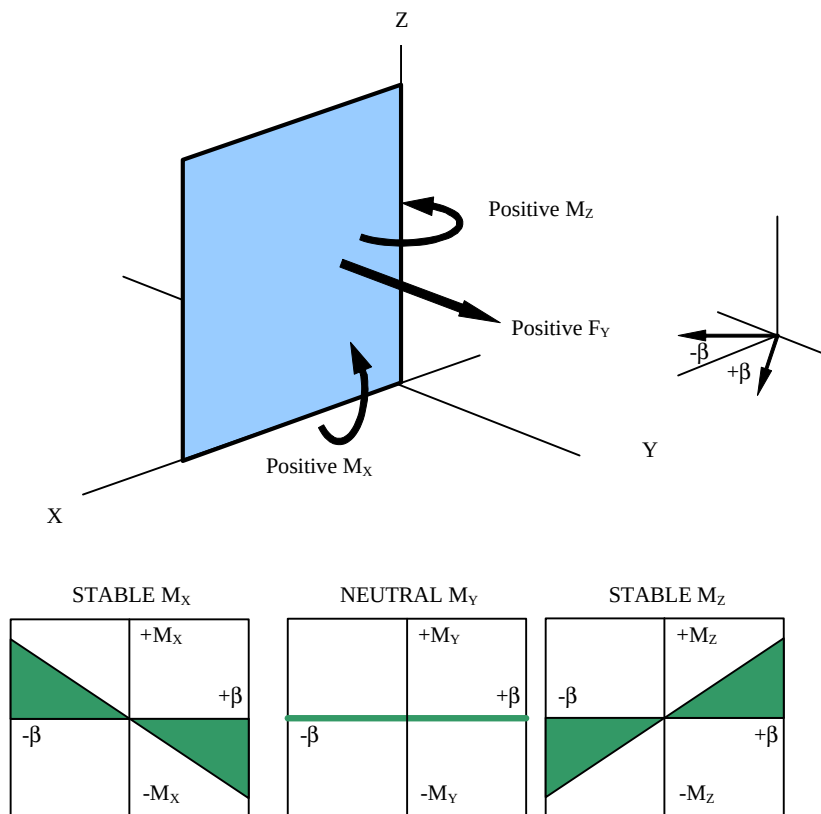
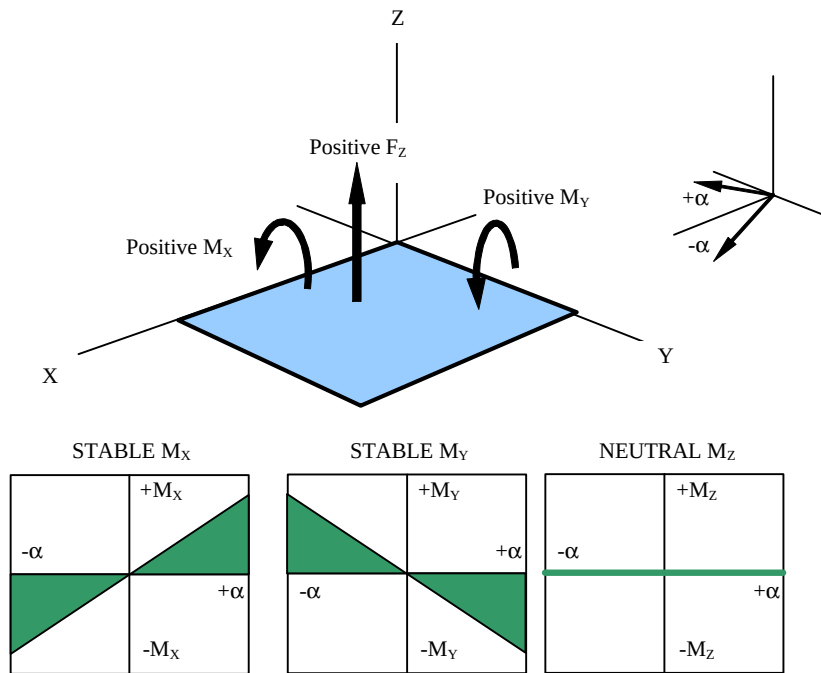
Will the computer refined planes always win? In general, no. In most weather conditions a thermal duration contest is still 90% pilot 10% airplane. The contest placings usually sort the pilots by skill regardless of what they are flying. If anyone wants to play with a vortex lattice program, contact me and I can email you directions on how to obtain a public domain program.

Reprinted from: <http://eiss.cnde.iastate.edu/articles/VortexLattice.htm>

NOTE: This article available online from the above link and is therefore assumed public (in the public domain). It was not written with SURFACES specifically in mind, but is reprinted here as the editor of this manual considered it well written and pertinent to anyone using CFD methods. Great OWL Publishing reprints it here for your convenience, but assumes no responsibilities for it.

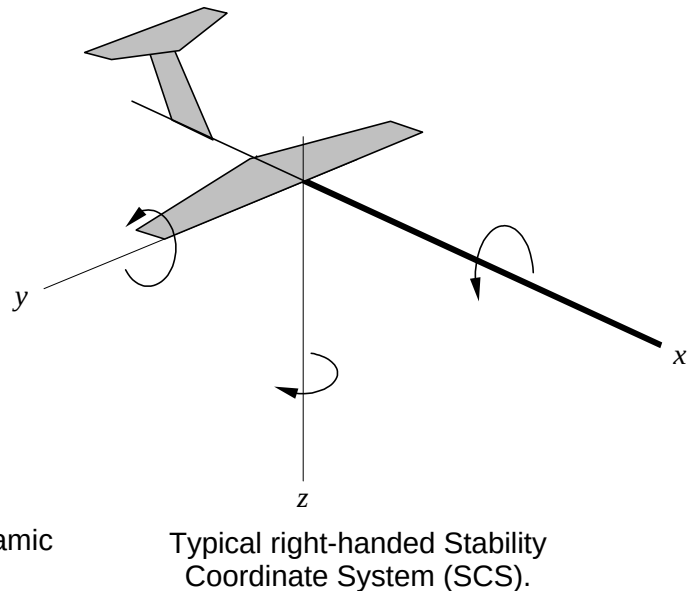
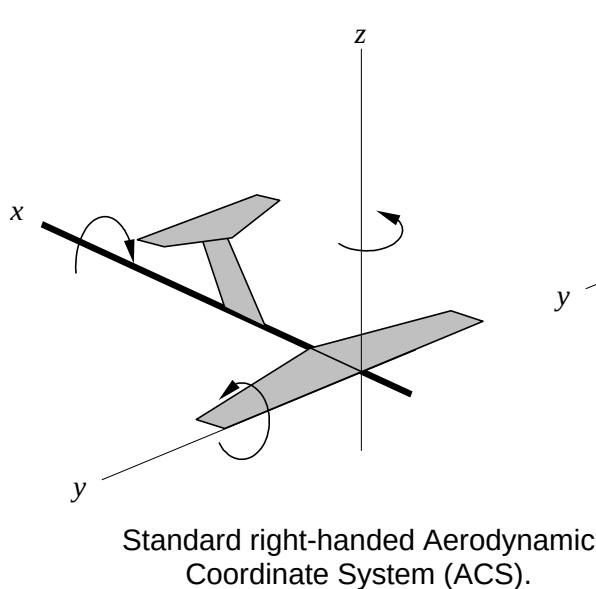
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## Orientation of Forces and Moments



## Force and Moment Nomenclature

| Name                                          | SURFACES Symbol | Other names |
|-----------------------------------------------|-----------------|-------------|
| Axial force (along X-axis)                    | FX              | X           |
| Side force (along Y-axis)                     | FY              | Y           |
| Normal force (along Z-axis)                   | FZ              | Z           |
| Rolling moment (about X-axis)                 | MX              | L           |
| Pitching moment (about Y-axis)                | MY              | M           |
| Yawing moment (about Z-axis)                  | MZ              | N           |
|                                               |                 |             |
| Coefficient of axial force (along X-axis)     | Cx              | $C_x$       |
| Coefficient of side force (along Y-axis)      | Cy              | $C_y$       |
| Coefficient of normal force (along Z-axis)    | Cz              | $C_z$       |
| Coefficient of rolling moment (about X-axis)  | Cl              | $C_l$       |
| Coefficient of pitching moment (about Y-axis) | Cm              | $C_m$       |
| Coefficient of yawing moment (about Z-axis)   | Cn              | $C_n$       |



Note 1: Positive rotation about an axis is always in the direction of the thumb of the right hand, as can be seen in the above figure.

Note 2: **SURFACES** uses a standard right handed Aerodynamic Coordinate System (ACS), which is conventionally used for other aspects of aircraft aerodynamic analyses. In this coordinate system, the sign of the lift is positive, when pointing upwards (i.e. towards positive Z), and the sign of the drag is positive, when pointing backwards (i.e. towards positive X). The user must be cognizant of the orientation of the axes when interpreting results.

Note 3: **SURFACES** comes with a routine that will convert stability derivatives to a standard body axes Stability Coordinate System (SCS). This is typically the default for stability and control related tools.

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## Project Task List

A typical project in **SURFACES** is conducted per the following list:

### Model Creation

| Task | Description     | Remark                                                                                                                                         |
|------|-----------------|------------------------------------------------------------------------------------------------------------------------------------------------|
| 1    | Define Points   | Drop points as required to represent the extremes of the aircraft.                                                                             |
| 2    | Define Vectors  | Draw vectors, parametric curves, or Bezier curves as needed, using the points. Use parametric or Bezier curves to represent cambered airfoils. |
| 3    | Create Surfaces | Define surfaces by selecting the opposite curves A1 and A2, and B1 and B2. Only use curves A1 and A2 for curved surfaces.                      |

### Model Preparation

| Task | Description                                        | Remark                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                                |
|------|----------------------------------------------------|---------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| 4    | Determine the Trapezoidal Mean Aerodynamic Chord   | <p>Select <b>Tools -&gt; Trapezoidal Mean Aerodynamic Chord...</b> from the VLM Console.</p> <p>This tool will determine several important geometric reference parameters to use with your model, including the MAC, its location, the wing area, and wing span. It also allows you to specify the CG location in terms of %MAC. You must use the <i>Transfer</i> tab on the form to transfer the calculated values to your model. While not necessary, it's recommended you copy the analysis report and paste as a Remark with your model. Do this by selecting <b>Edit -&gt; Remark...</b> from the Surfaces Worksheet window.</p> |
| 5    | Determine the Horizontal and Vertical Tail Volumes | <p>Select <b>Tools -&gt; Horizontal/Vertical Tail Volume...</b> from the VLM Console.</p> <p>Although not necessary for analysis, it is a good idea to tail volume and compare to other airplanes. Copy and paste the analysis report into the remark.</p>                                                                                                                                                                                                                                                                                                                                                                            |

|   |                        |                                                                                                                                                                                                                       |
|---|------------------------|-----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| 6 | Other model properties | <p>Select <b>Edit -&gt; Model Properties...</b> from the VLM Console.</p> <p>Try to fill in as many properties as possible. All entries marked with an asterisk (*) are required for any Vortex-Lattice analyses.</p> |
|---|------------------------|-----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|

Once your model runs, you can initiate a large number of specific investigations.

## Basic Investigations

| Task | Description             | Remark                                                                                                                                                                                                                                                                                                                                                                                                           |
|------|-------------------------|------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| 7    | Determine Neutral Point | <p>Select <b>Tools -&gt; Determine Neutral Point...</b> from the VLM Console.</p> <p>This is a necessary step as it will determine your aft CG limit. Always consult the CG location of your design with a qualified Aerospace Engineer. The CG is typically at least 8-10% MAC forward of the neutral point.</p>                                                                                                |
| 8    | Trim Analysis           | <p>Select <b>Tools -&gt; Determine Neutral Point...</b> from the VLM Console.</p> <p>This tool is helpful to determine required surface deflections for given weights, airspeed, and yaw angles. Note that before you can use this tool, you must define control surfaces using edge deflections and proper references under the Edit Surface dialogbox (<i>Edge Deflections</i> and <i>Reference</i> tabs).</p> |
| 9    | Panel Results           | <p>Select the <i>Panel Results</i> tab on the VLM Console.</p> <p>Here you can extract various information pertaining to panels, such as areas, normals, vortex strengths, velocity over a panel, force generated by a panel, pressure coefficients, panel lift coefficients, as well as the center of pressure.</p>                                                                                             |
| 10   | Body Results            | <p>Select the <i>Body Results</i> tab on the VLM Console.</p> <p>Here you can extract information about forces and moments acting on your model.</p>                                                                                                                                                                                                                                                             |

|    |               |                                                                                                                                                                                                                                                                                               |
|----|---------------|-----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| 11 | Strip Results | <p>Select the <i>Panel Results</i> tab on the VLM Console.</p> <p>Here you can extract a number of information about strips of panels (chordwise), for instance, forces, moments and coefficients. Display strip CL (section lift coefficients) to help you design for delayed tip stall.</p> |
|----|---------------|-----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|

You can conduct even more sophisticated analysis per the following task list.

## Advanced Investigations

| Task | Description                                    | Remark                                                                                                                                                                                                                                                                                                                                                                                     |
|------|------------------------------------------------|--------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| 12   | Determine Stability Derivatives                | Select <b>Tools -&gt; Determine Stability Derivatives...</b> from the VLM Console.                                                                                                                                                                                                                                                                                                         |
| 13   | Determine Control Response                     | Select <b>Tools -&gt; Determine Control Response...</b> from the VLM Console.                                                                                                                                                                                                                                                                                                              |
| 14   | Determine Loads                                | Select <b>Results -&gt; Force Integrator...</b> from the VLM Console.                                                                                                                                                                                                                                                                                                                      |
| 15   | Determine Specific Conditions                  | <p>Select <b>Tools -&gt; Goal Seek...</b> from the VLM Console.</p> <p>With this tool you can calculate AOA, AOY, or Vinf required to generate a specific load, lift, or even lift coefficient. Note the result don't necessarily result in an aerodynamically balanced model (i.e. MX, MY, or MZ will be non-zero).</p>                                                                   |
| 16   | Modify Geometry to Satisfy Specific Conditions | <p>Select <b>Tools -&gt; Geometric Goal Seek...</b> from the VLM Console.</p> <p>This tool can be used to move points so that specific conditions are satisfied. The best example of its use is to move the leading points on a stabilator in the Z-directions at a specific flight condition so the MY is zero. In other word, determine an ideal angle of incidence of a stabilator.</p> |
| 17   | Virtual Wind Tunnel                            | Select <b>Virtual WT -&gt; Setup and Execute WT Run...</b> from the VLM Console.                                                                                                                                                                                                                                                                                                           |

## Creating a Simple Model with SURFACES

The following model is designed to allow the novice user a very quick introduction to **SURFACES**. As a reminder, pay close attention to which options and checks are made in each form below.

STEP 1: Start a new project. Select File->New...

STEP 2: Select Insert->Trapezoidal Surface...

STEP 3: Create the **WING** using the numbers in the dialog in Figure 1.

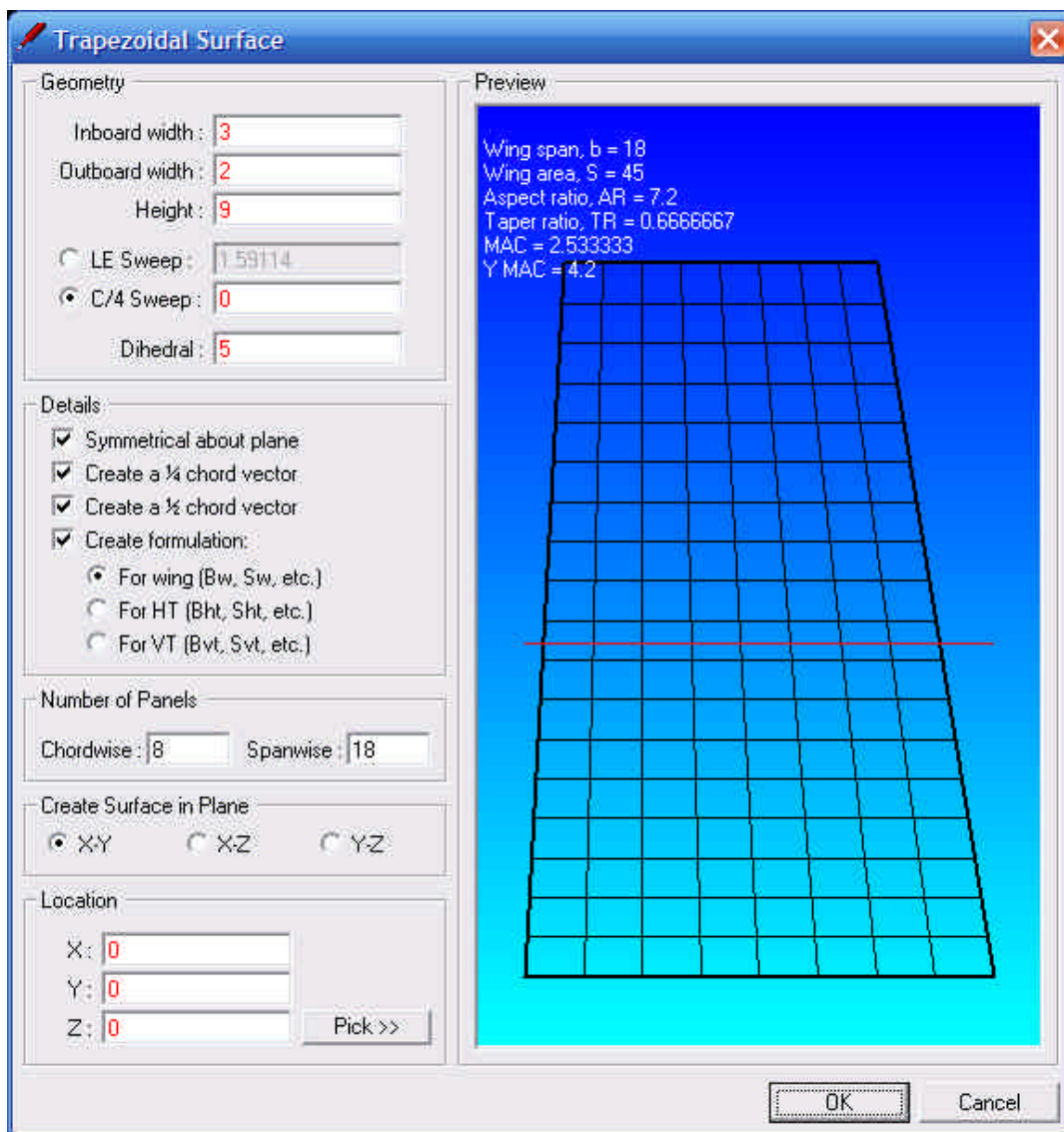


Figure 1: Creating the wing.

STEP 4: Create the **HORIZONTAL TAIL** by filling in using the numbers in the dialog in Figure 2.

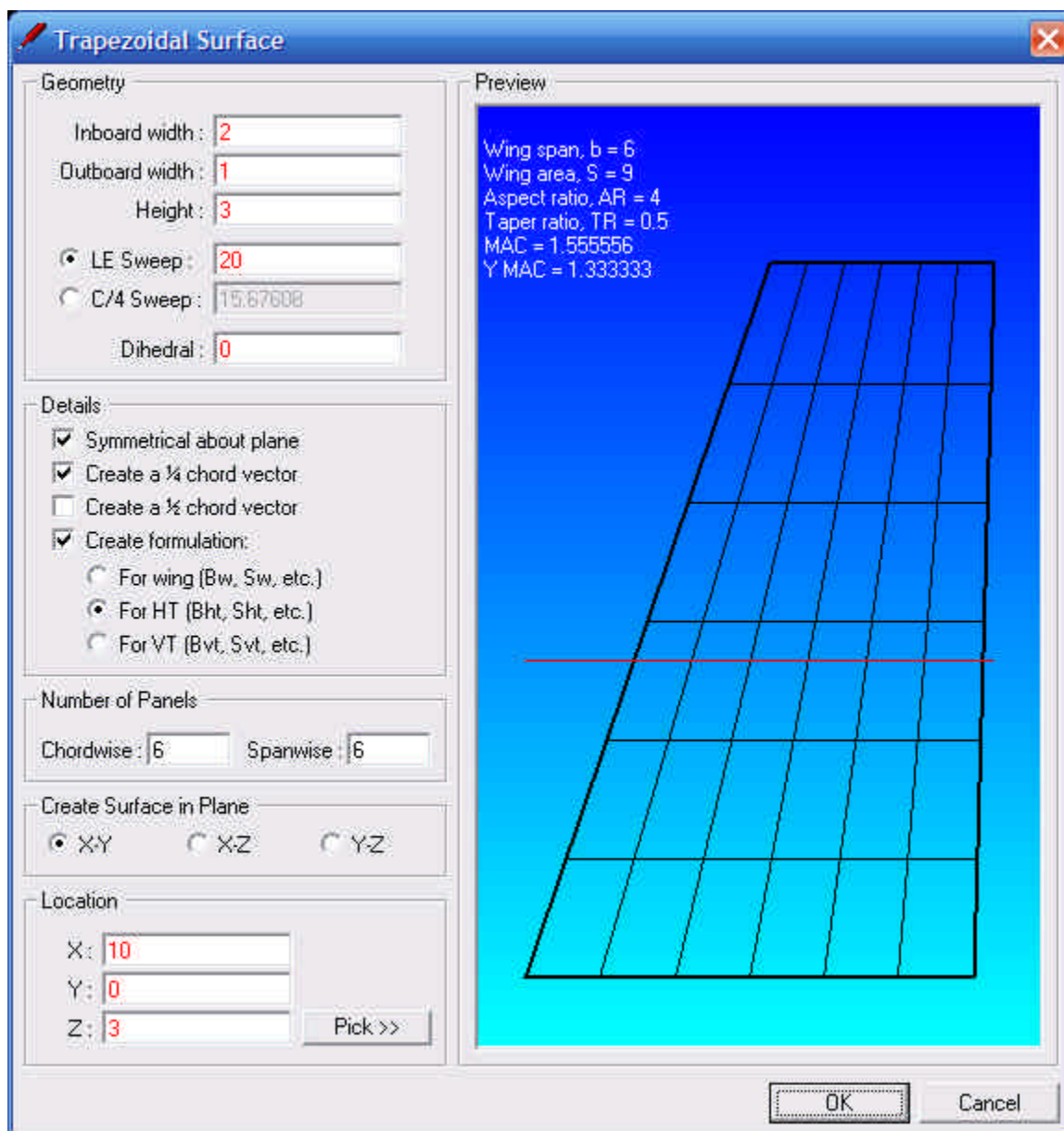


Figure 2: Creating the HT.

STEP 5: Create the **VERTICAL TAIL** by filling in using the numbers in the dialog in Figure 3.

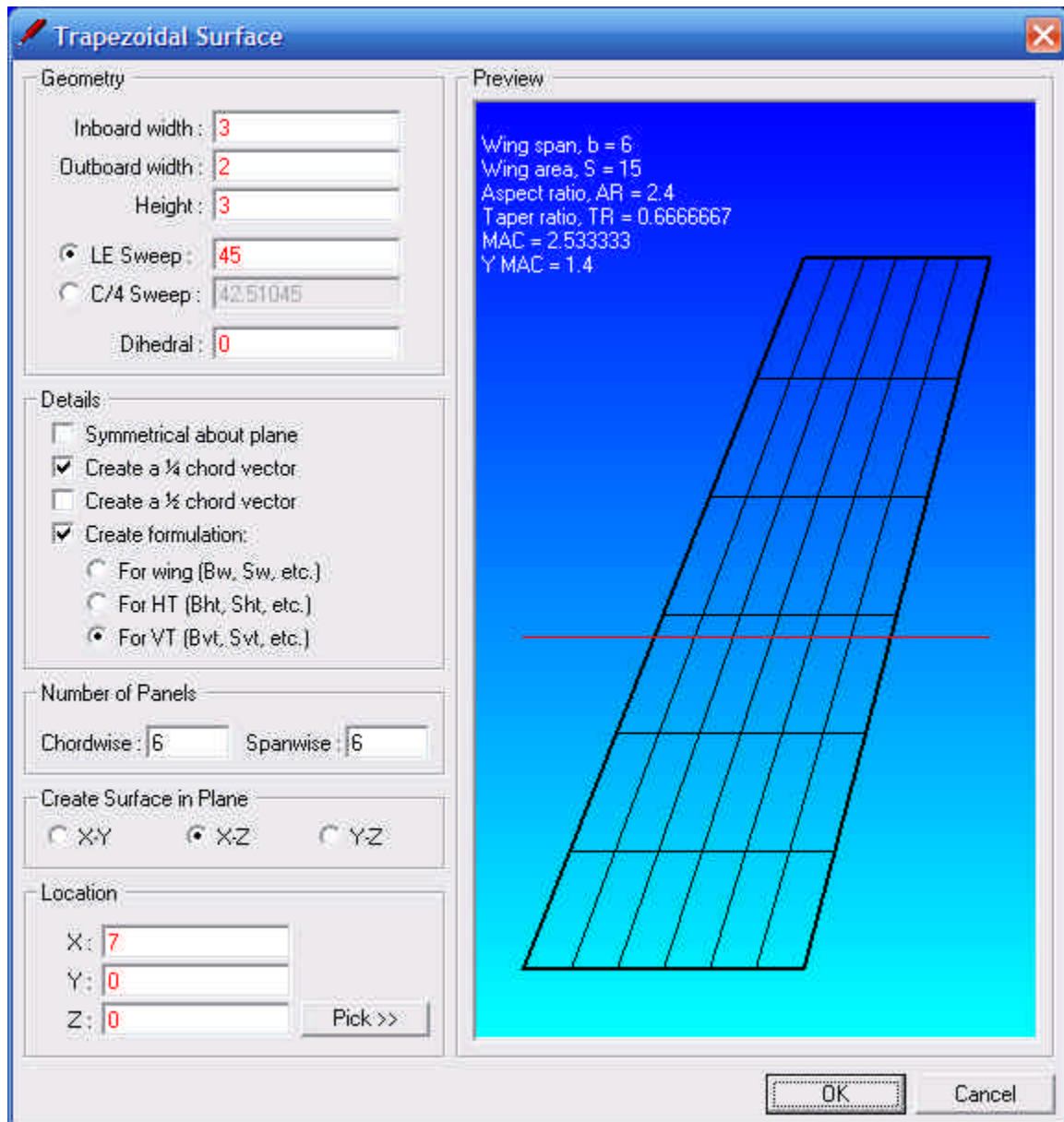


**Figure 3: Creating the VT.**

When complete your model should look like the one in Figure 4; a T-tail design with a straight tapered wing. You should be aware of that you can also create the surfaces directly by dropping points, stretching vectors, and inserting surfaces. However, in the interest of speed, the user can create trapezoidal surfaces more easily using this tool.

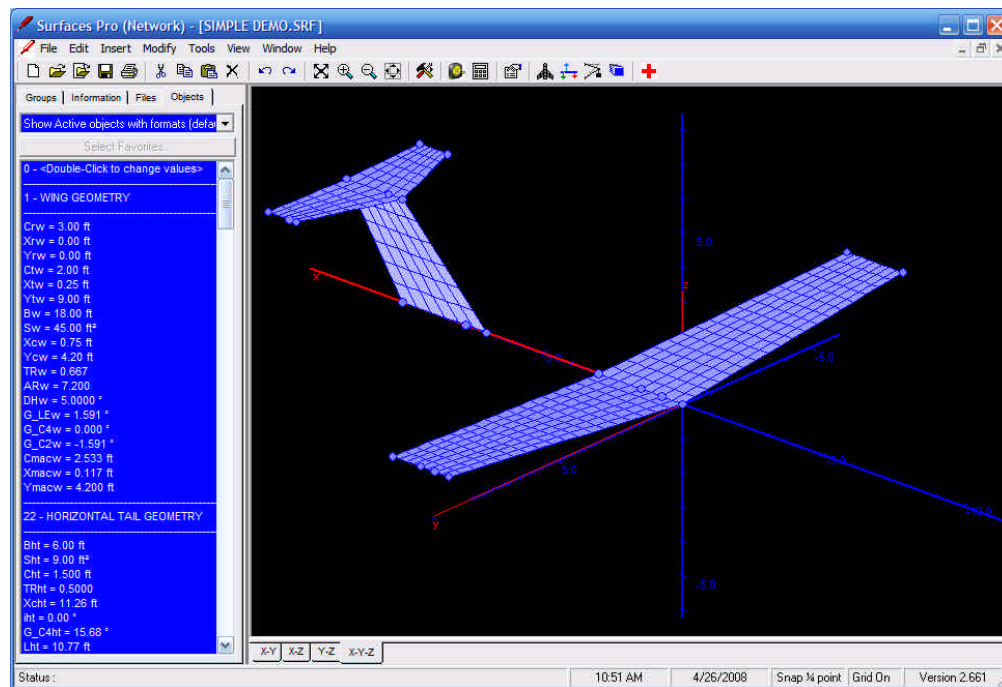


**Figure 4: Completed basic model.**

The wing span ( $B_w$ ) is 18 ft, wing area ( $S_w$ ) is 45 ft<sup>2</sup>, and the horizontal and vertical tail volumes should be 0.8504 and 0.0827, respectively. Now let's add weight to the model using the specialized tools in **SURFACES**.

**STEP 6:** Select Edit->Select Surfaces... Press the [Select All] button and then the [OK] button.

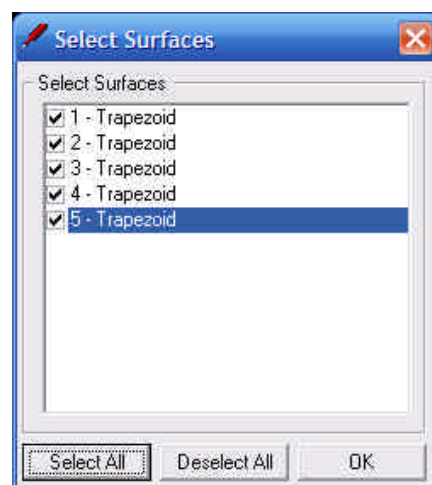


**Figure 5: Selecting all surfaces simultaneously.**

**STEP 7:** Select Tools->Distribute Weight on Selected Surfaces and Nodes... Enter 400 in the entry box and press the [OK] button.

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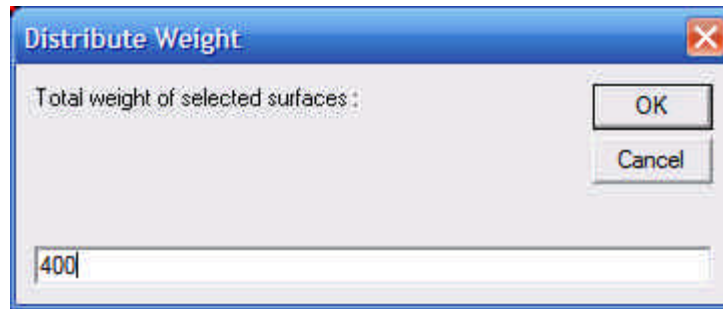


**Figure 6: Enter weight of the selected surfaces, here as 400 lbs.**

This will distribute a total weight of 400 lbs onto the model based on the area. That is, **SURFACES** calculates the total area of the selected surfaces and then computes weight per total area. The weight property of each surface will then be assigned a weight, which is calculated as (weight per total area) x (surface area). As a consequence, the total weight of the wings is 293 lbs, the HT weighs 58.4 lbs, and the VT weighs 48.6 lbs. Clearly, this adds up to 400 lbs.

**STEP 8: Make sure the CG is visible. Select Tools->Options... Check the 'Show CG, Neutral Point, Aerodynamic Center' checkbox and press the [OK] button.**

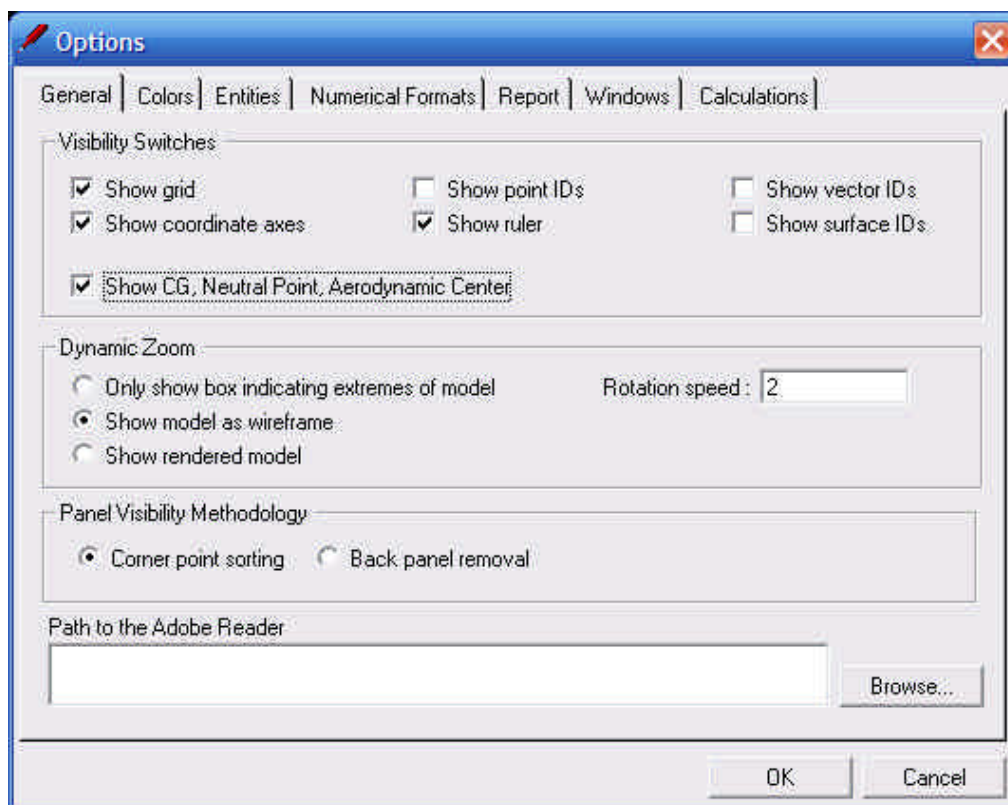


**Figure 7: Make sure the CG checkbox is marked so you can see the CG in the workspace.**

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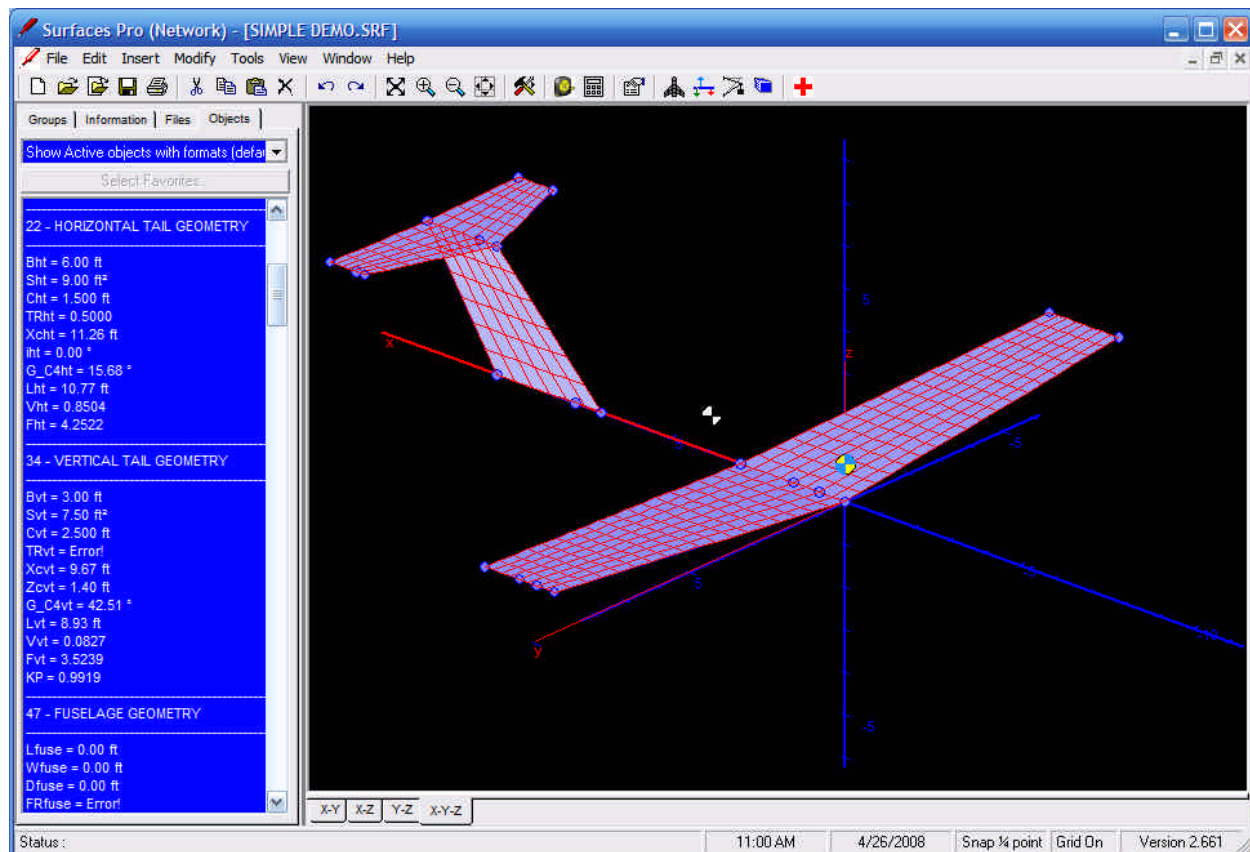
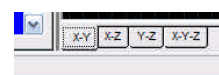


**Figure 8: We can see the CG location (black-white circle) is way aft.**

When completed your model should look as the one in Figure 8. It is immediately evident that the CG is too far aft. Therefore, let's create a ballast point to allow us means to move it around.

**STEP 9:** Press the X-Y tab on the bottom of the workspace. This will display the model projected onto the X-Y plane.



**STEP 10:** Press the sketchmode icon to display the sketch toolbar.



**STEP 11:** Press the Insert Point icon and drop a point somewhere in front of the wing, near the X-axis, similar to what is shown in Figure 9.



**STEP 12:** Select the point by clicking on it and press the Insert Node icon to convert it to a node. This will open a dialog box to allow user to enter additional data. Enter the information shown in Figure 10. Once completed, press the [OK] button.



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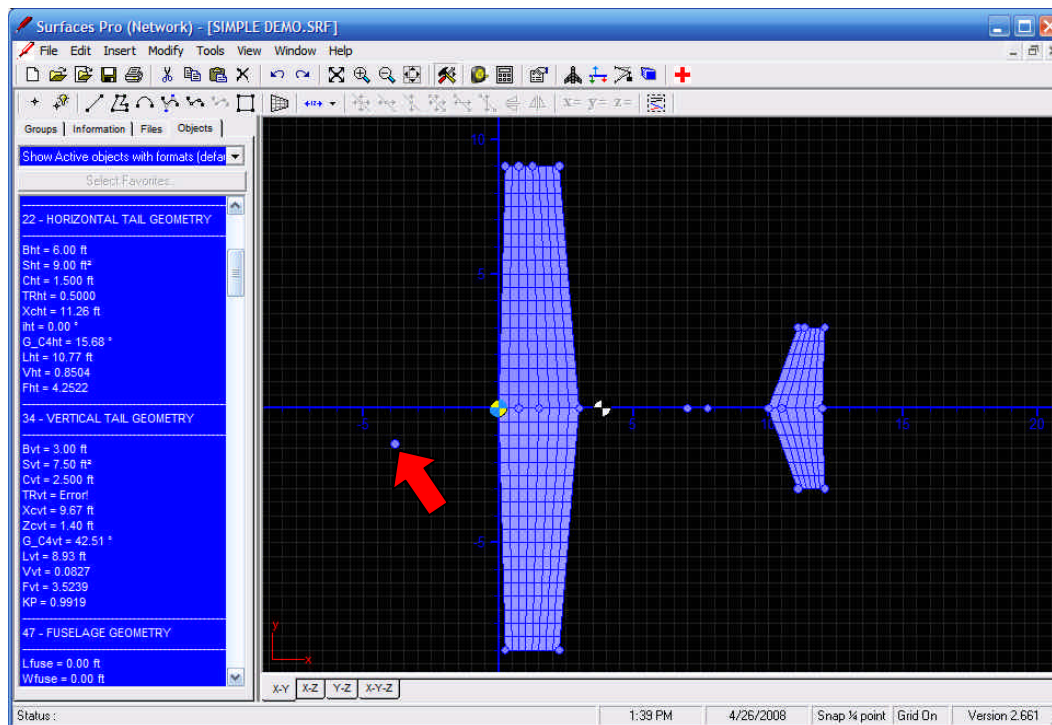


**Figure 9: Drop the point (to be converted to a node) in a location similar to the one shown.**

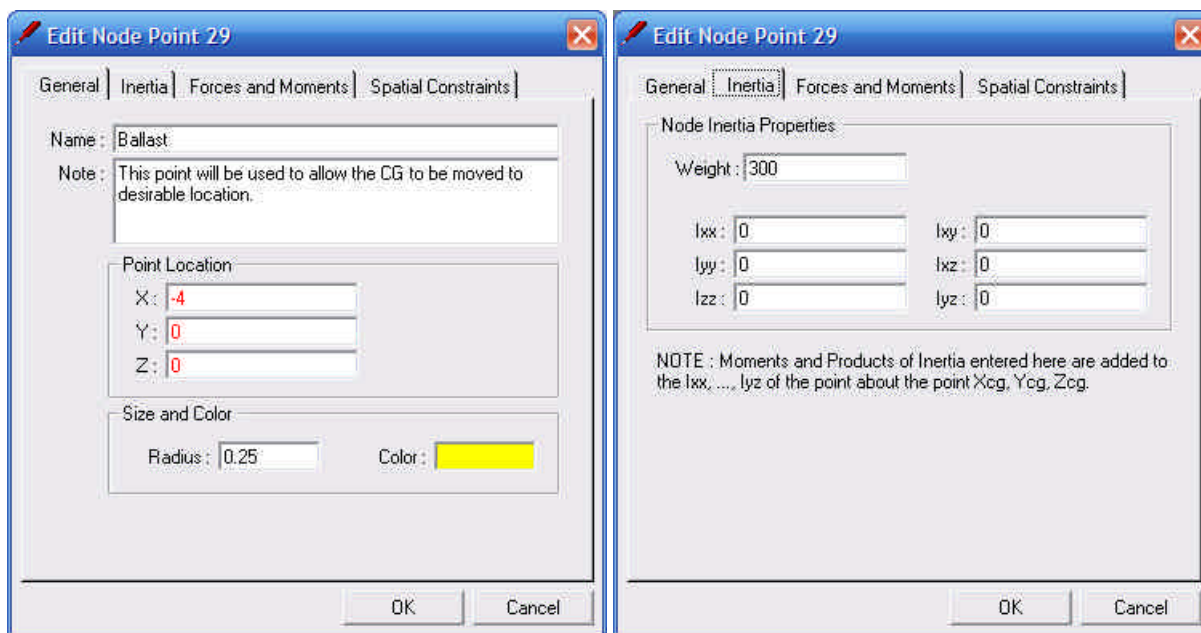


**Figure 10: Information entered with Step 12.**

When completed your model should look as the one in Figure 11. To see what the true location of the CG is at this point, locate the Xcg math object in the object list on the left hand side (highlighted in Figure 11). The variable Pmac stores the CG location as a percentage of the Mean Aerodynamic Chord (Cref,

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obtained in the REFERENCE PARAMETERS block above). We see the CG is located at 14.181% MAC or at 0.48 ft. Often we prefer to specify directly the location of the CG. **SURFACES** comes with a tool to help with that. The following steps show how to move the CG to 25% MAC.

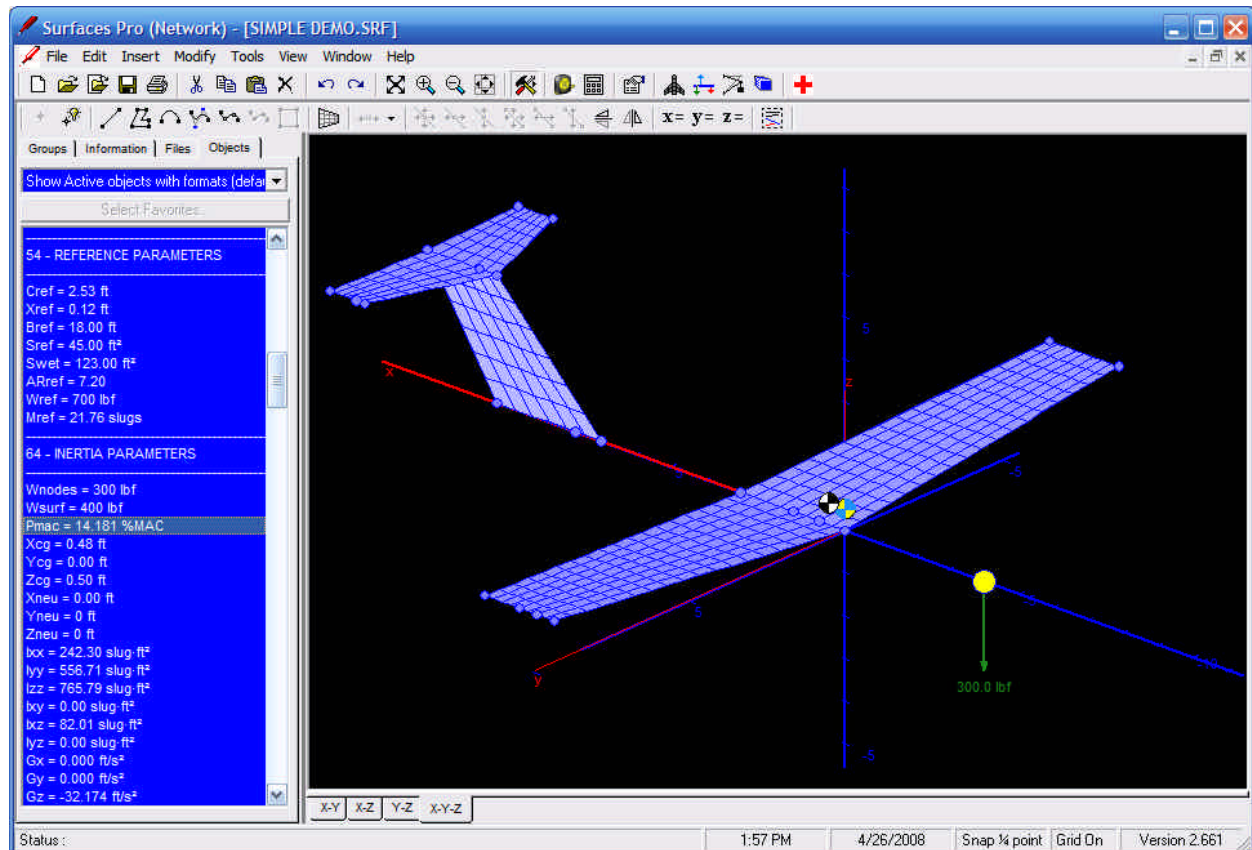


**Figure 11: The model with ballast point defined.**

STEP 13: Click once on the Ballast node to select it. Note that **SURFACES** will only move the selected node or nodes, when adjusting the CG location. If none are selected a warning message appears.

STEP 14: Select Tools->Specify a CG Location... Select the option and enter the value shown in Figure 12.

STEP 15: Press the [Adjust] button. Respond to the warning that appears by pressing [Yes]. Then, press [Close] button to exit the form.

When completed your node will appear closer to the wing than before, or but **SURFACES** has automatically changed its X location from -4 to -3.360483 ft, moving the CG in the process (i.e. to the 25% MAC). Now let's learn some more details about the model. Let's determine the neutral point per the following steps.

STEP 16: Press the VLM Console icon. This will open the Vortex-Lattice Method Console shown in Figure 13.



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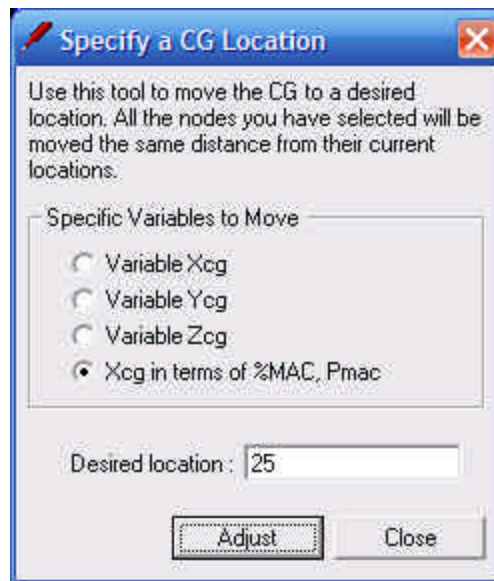


Figure 12: Specifying a CG location.

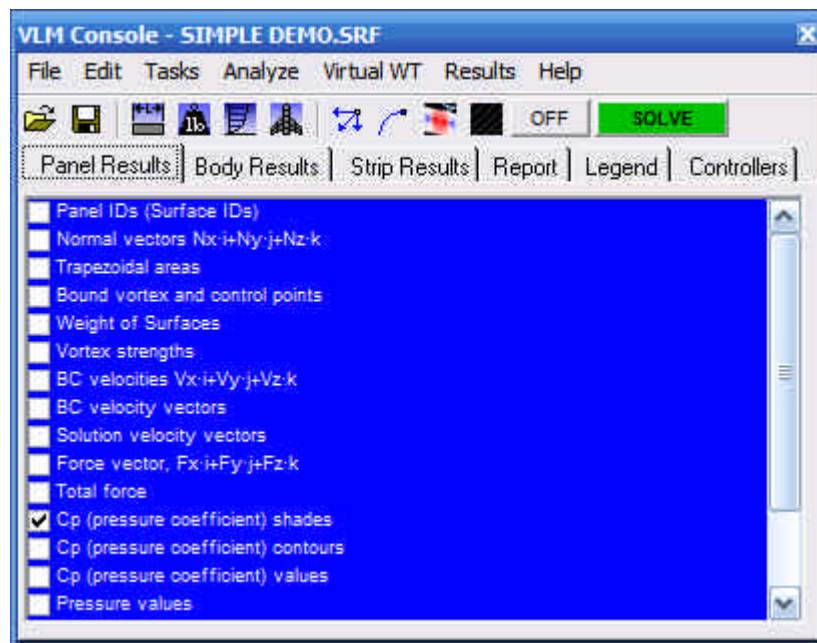


Figure 13: The VLM Console.

Note that when you create a new project, **SURFACES** has preset values for a multitude of variables. Among those are the airspeed variables ( $V_{cas}$ ,  $V_{tas}$ ,  $V_{inf}$ ), altitude ( $H_{ref}$ ), and angle-of-attack ( $AOA$ ). Naturally, you can change these with ease, but currently  $V_{cas}=100$  knots,  $H_{ref}=0$  ft, and  $AOA=2^\circ$ . In interest of saving time for this demo, let's assume these will suffice for our analysis.

**STEP 17:** Select **Tasks->Determine Neutral Point...** Press the [Analyze] button to begin, and after a few seconds, once done, review the results in Figure 14.

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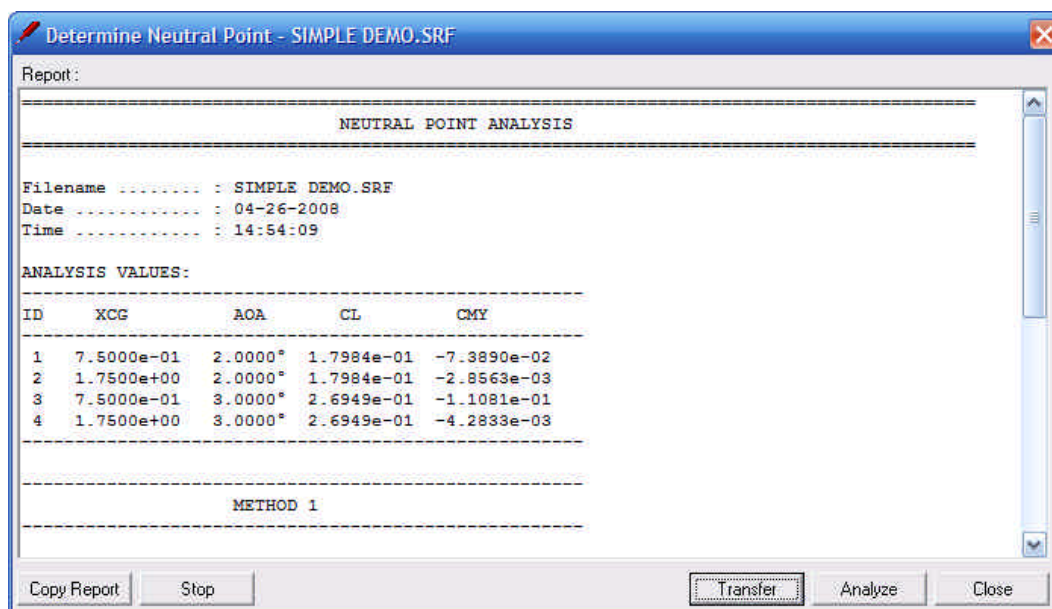


**Figure 14: Determining neutral point.**

The full report is displayed below. Note that **SURFACES** uses two methods to compute the neutral point. Generally you should pick the neutral point with the lower value of Xneu, here this implies Method 2. Let's transfer the resulting value to the variable Xneu in the model, which currently has the initial value 0.

```
=====
                        NEUTRAL POINT ANALYSIS
=====

Filename ..... : SIMPLE DEMO.SRF
Date .....     : 04-26-2008
Time .....     : 15:47:48

ANALYSIS VALUES:
-----
ID    XCG      AOA      CL      CMY
-----
1    7.5000e-01  2.0000°  1.7984e-01  -7.3890e-02
2    1.7500e+00  2.0000°  1.7984e-01  -2.8563e-03
3    7.5000e-01  3.0000°  2.6949e-01  -1.1081e-01
4    1.7500e+00  3.0000°  2.6949e-01  -4.2833e-03
-----

                        METHOD 1
-----

Calculates Xneu from the expression:

      Xneu = Xcg - Cref·dCm/dCL
            = 0.7499992 - (2.533333)·(-0.036917)/(0.089648)
            = 1.793228 (66.18005% MAC)

-----

                        METHOD 2
-----

Calculates Xneu by evaluating changes of CG and AOA on Cm:

Function 1 (degrees): -0.036917·AOA - 0.000056
Function 1 (radians): -2.115189·AOA - 0.000056
```

```
Function 2 (degrees): -0.001427·AOA - 0.110807
Function 2 (radians): -0.081765·AOA - 0.110807

Xneu = 1.79021 (66.06091% MAC)

Total time:0h:00m:05s
```

STEP 18: Press the [Transfer] button and select the option 'Neutral point using Method 2'. This displays a notification. Press the [OK] button to close it. Press the [Close] button on the form to close it as well.

It is a good practice to copy the entire report and paste it as a comment under Edit->Remark... in the main worksheet. Now lets trim the aircraft for a level flight. First we must define which surfaces serve as the elevators.

STEP 19: Double-click on first of the two smaller horizontal surfaces. This opens the dialog box shown in Figure 15. Select the 'Edge Deflections' tab. Set number of chordwise panels on the aft edge to deflect to 2.

STEP 20: In the same dialog select the 'Reference' tab. Check the 'Surface is used for Pitch Control'. Press the [OK] button.

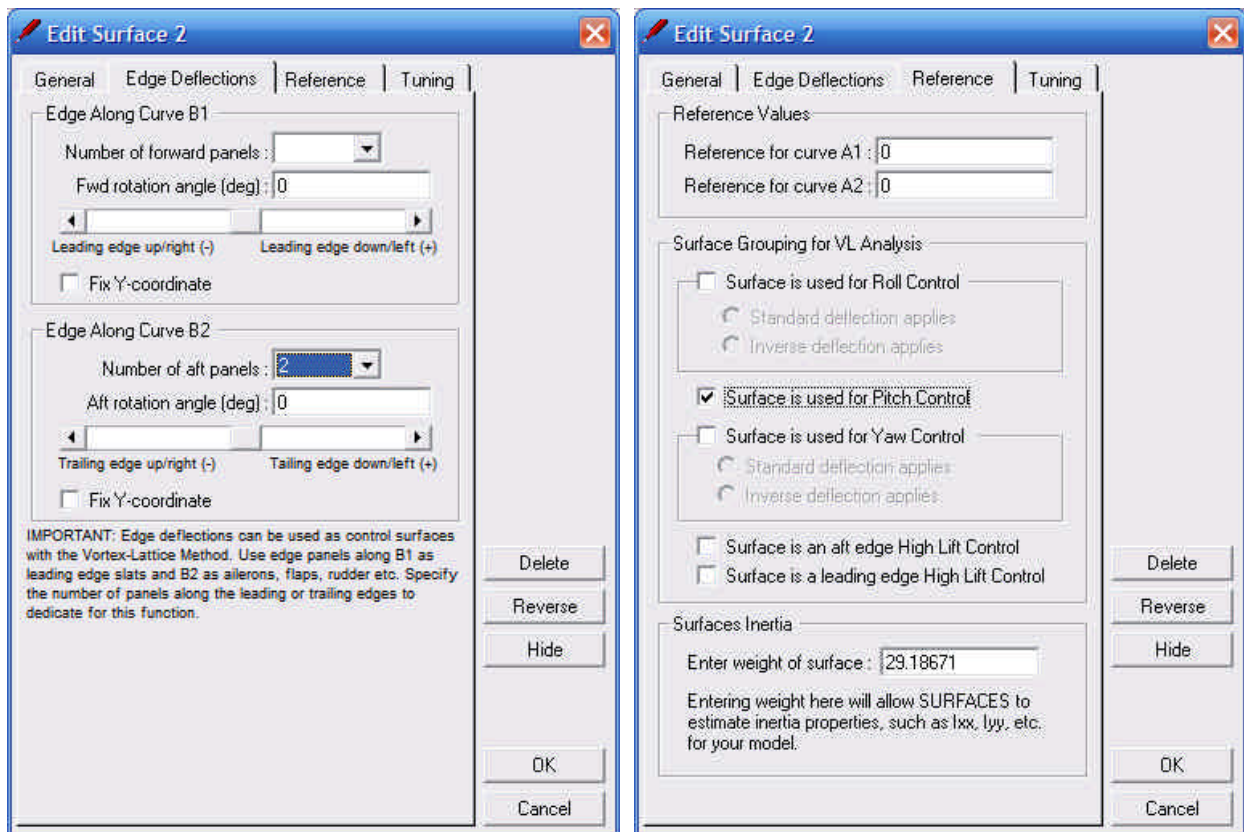


Figure 15: Setting up elevator functionality.

STEP 21: Repeat Steps 19 and 20 for the other small horizontal surface.

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You have now given **SURFACES** information it can use to automatically deflect the elevators to trim the model for level flight. You can try the functionality out by displaying the VLM Console and select the 'Controllers' tab. Move the Pitch control scroll bar to see the model regenerate with the desired deflection as shown in Figure 16. Once done, press the [Reset] button to return the elevators to a neutral deflection (0°) and get ready to trim the model.

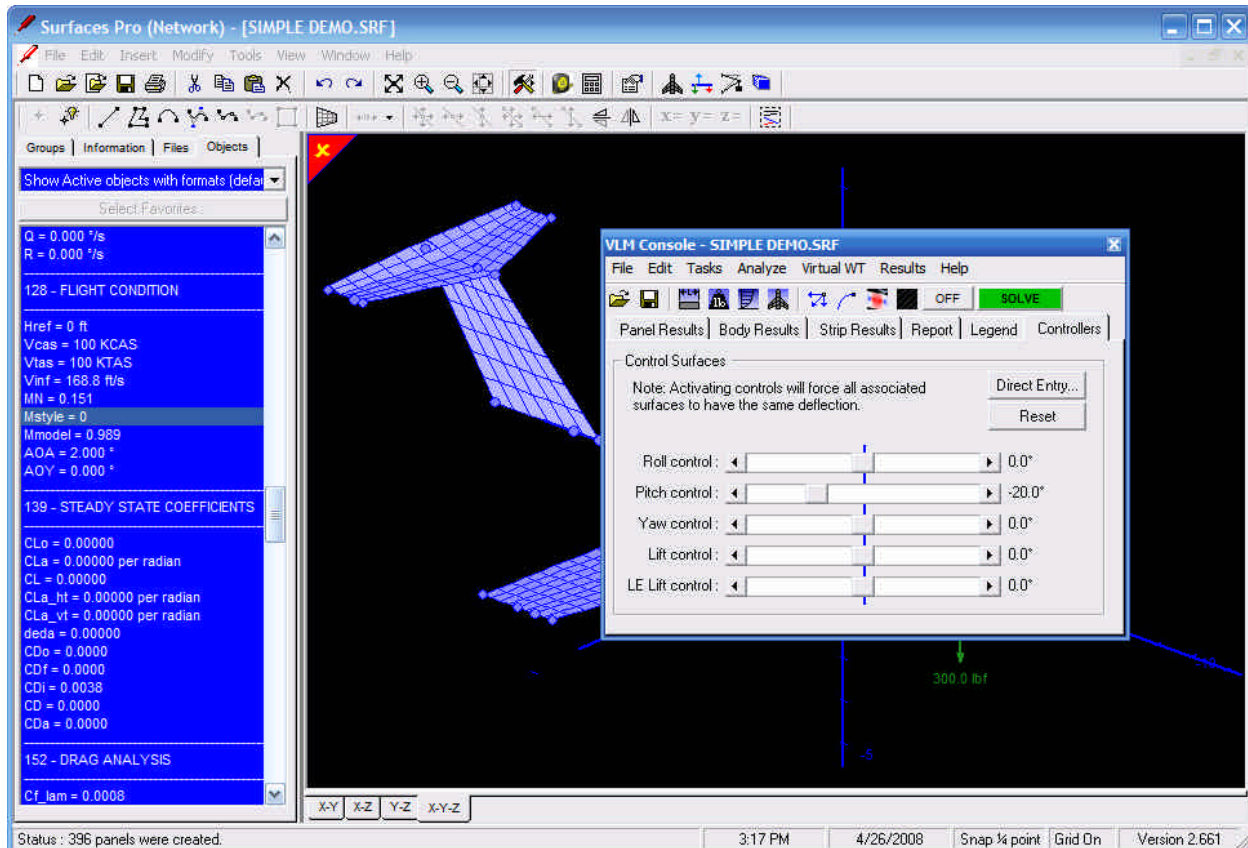
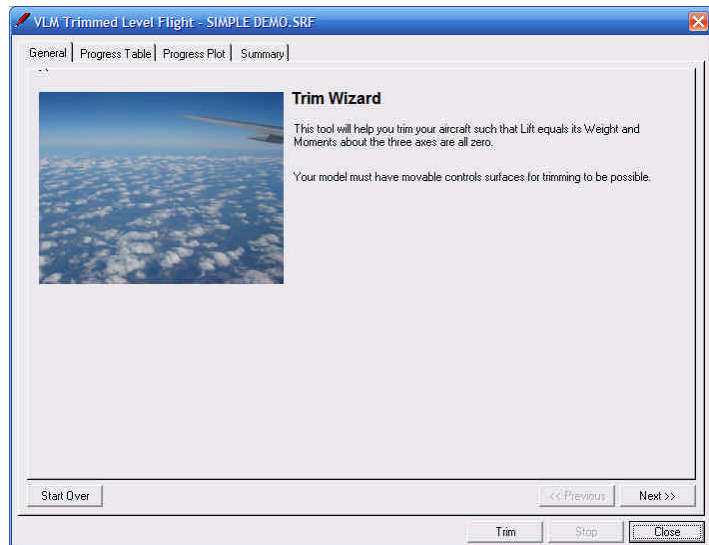


**Figure 16: Demoing elevator functionality.**

STEP 22: Select Tasks->Trimmed Level Flight... to display the Trim wizard. Follow the steps shown in the subsequent list of images.

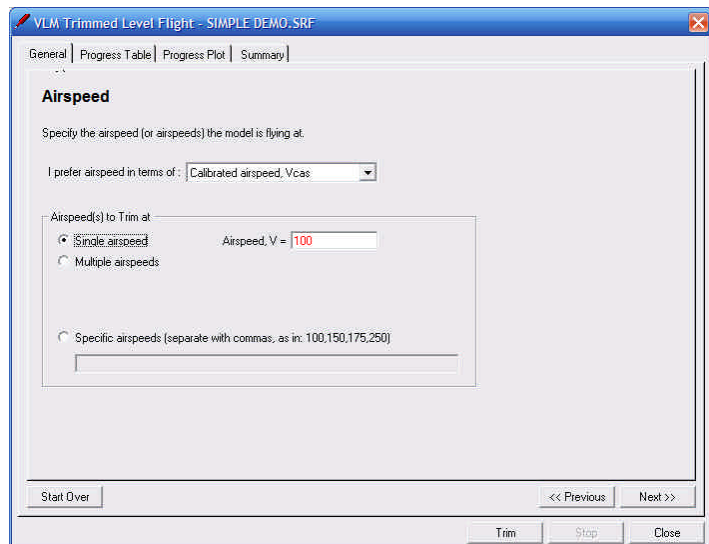
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STEP 22a:  
Press the [Next >>] button.



STEP 22b:  
Ensure the selection shown.  
Press the [Next >>] button.

We will just trim to a single  
airspeed, but multiple  
airspeeds can also be analyzed.



STEP 22c:  
Ensure the selection shown.  
Press the [Next >>] button.

Once complete, the lift  
generated will be 700 lbs at  
the airspeed specified in Step  
22b.

**VLM Trimmed Level Flight - SIMPLE DEMO.SRF**

General | Progress Table | Progress Plot | Summary

**Flight Condition**

Specify weight and yaw attitude of the model (angle-of-attack will be determined) and axes to trim about.

The initial weight is always the current weight of the model, but you can change it. Take advantage of the fact that you can enter an arithmetic expression in the weight box. For instance, if you model weighs 3000 lbs and you want to determine loads at 3.8gs, simply type '3.8\*3000' and hit Enter.

Analysis Targets:

Weight : 699.9999

Angle-of-Yaw : 0

Select which axes the model will be trimmed about. Your model must have ailerons to trim about the X-axis (roll), elevator to trim about the Y-axis (pitch), and rudder to trim about the Z-axis (yaw).

Axes to Trim About:

☐ Trim about X-axis

☒ Trim about Y-axis

☐ Trim about Z-axis

Start Over << Previous Next >>

Trim Stop Close

STEP 22d:  
Ensure the selection shown.  
Press the [Next >>] button.

Here we allow 30 iterations  
before a solution will be  
declared as unachievable. If  
solution is found, the  
resulting lift will be 700 ±1  
lb<sub>f</sub> and the moment 0 ±1 ft·lb<sub>f</sub>.  
As a rule of thumb, acceptable  
accuracy is provided by  
specifying 1% of the weight.  
Here, the accuracy is closer to  
0.14%.

**VLM Trimmed Level Flight - SIMPLE DEMO.SRF**

General | Progress Table | Progress Plot | Summary

**Analysis Specifics**

Here specify details for the trim algorithm. Lift and moment tolerances are used to determine how close to weight the lift is and how close the moment is to zero before trimming is considered complete. The smaller the tolerance the closer the final values will be to target values, but more time will be required.

Analysis Specifics:

Maximum number of iterations : 30

Lift tolerance : 1

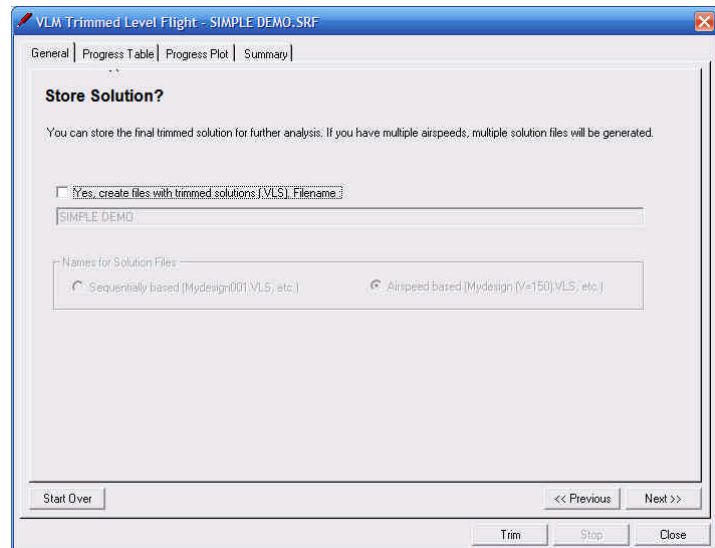
Moment tolerance : 1

Start Over << Previous Next >>

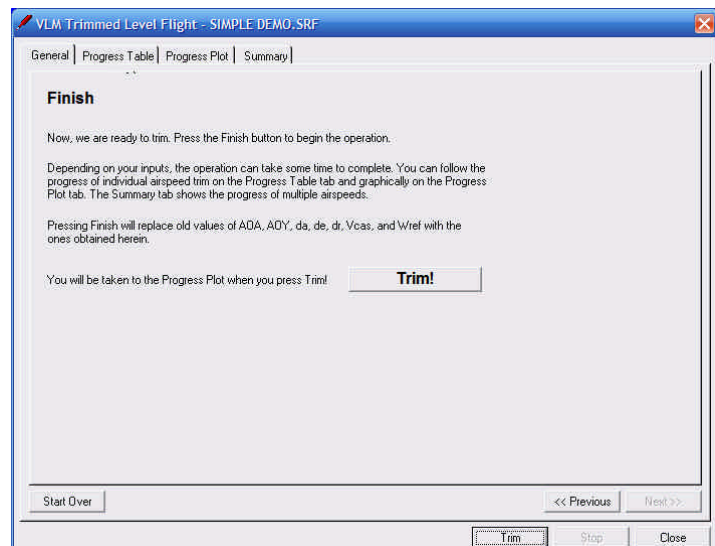
Trim Stop Close

STEP 22e:  
Ensure the selection shown.  
Press the [Next >>] button.

Note that solution files can be created and saved using the file name entered as a seed.



STEP 22f:  
Press the [Trim!] button.



Once **SURFACES** begins to trim, you can follow the progress on the 'Progress Table' or 'Progress Plot' tabs (see Figure 17). The time to trim largely depends on the number of panels in the model and accuracy desired. The model presented here took 16 iterations and 31 seconds to trim. Press 'Summary' tab to read the results for each completed trim speed. In this case, the model will fly level at an AOA of  $5.7137^\circ$  and will require an elevator deflection of  $-5.5623^\circ$  (trailing edge up) to balance. The lift generated is 699.011 lbf and moment about the y-axis (located at the CG) is 0.122438 ft-lbf. The model is automatically set to the resulting AOA and elevator deflection.

Next let's determine stability derivatives for the model in this particular configuration.

STEP 23: Select Tasks->Determine Stability Derivatives... to display the Stability Derivatives form. Check the boxes shown in Figure 18 and press the [Analyze] button.

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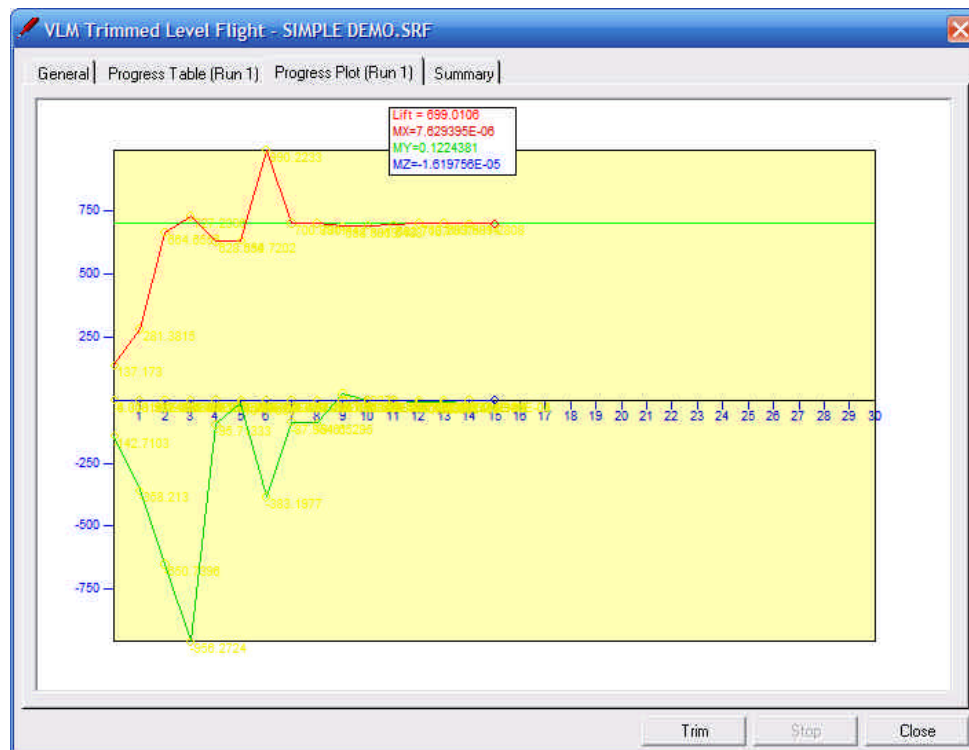


**Figure 17: Trim progress is displayed on the 'Progress Plot' tab.**

When the analysis completes you will see the results as shown in Figure 19. Without going into too many details, we can see from values for  $C_{ma}$  (-2.119),  $C_{lb}$  (-0.105), and  $C_{nb}$  (0.172) that our airplane is statically stable about all three axes. What we don't know at this time are its dynamic stability properties. And this is what we intend to investigate next. First, however, we must transfer these results to the airplane model.

STEP 23: Select the 'Transfer' tab. Follow the remaining steps closely.

STEP 24: Press the [Select All] button to select all the derivatives in the list.

STEP 25: Press the [Deselect Nonrequested] button to deselect the derivatives that were not calculated.

STEP 26: Uncheck the following variables:  $CL$ ,  $CD$ ,  $C_{da}$ ,  $hcg$ , and  $hn$  (see Figure 20). This will prevent them from being overwritten, but they already consist of algebraic expressions that we don't want to be deleted.

STEP 27: Press the [Transfer] button. Press the [OK] button on the notification form that pops up stating a successful transfer. Then press the [Close] button to close the Stability Derivatives form.

Now we proceed to the dynamic stability analysis.

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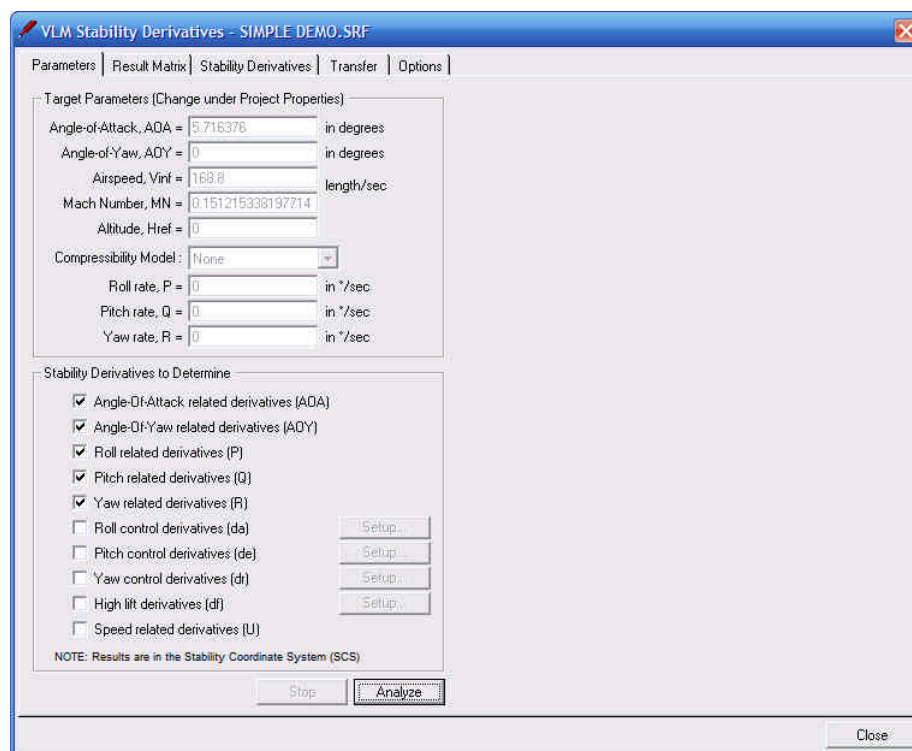


Figure 18: Preparing to determine stability derivatives.

|                                |      | Per degree | Per radian |
|--------------------------------|------|------------|------------|
| Angle-of-Attack                | ADA  | 5.7164 *   | 0.0997696  |
| Angle-of-Yaw                   | ADY  | 0.0000 *   | 0.0000000  |
| Far field speed                | Vinf | 168.8      | 168.8      |
| Density                        | rho  | 0.002378   | 0.002378   |
| Density altitude               | Href | 0          | 0          |
| Roll rate                      | P    | 0          | 0          |
| Pitch rate                     | Q    | 0          | 0          |
| Yaw rate                       | R    | 0          | 0          |
| Basic lift coefficient         | CLo  | -0.046144  | -0.046144  |
| Lift coefficient               | CL   | 0.458669   | 0.458669   |
| Lift curve slope               | CLa  | 0.088310   | 0.059786   |
| Induced drag coefficient       | CDi  | 0.011043   | 0.011043   |
| Drag coefficient               | CD   | 0.011043   | 0.011043   |
| Drag coefficient slope         | CDa  | 0.004213   | 0.241412   |
| FX variation with ADA          | Cxa  | 0.000092   | 0.005259   |
| FY variation with ADA          | Cya  | 0          | 0          |
| FZ variation with ADA          | Cza  | -0.089577  | -5.132387  |
| Rolling Moment wrt ADA (CMXA)  | Clx  | 0          | 0          |
| Pitching Moment wrt ADA (CMYA) | Cmx  | -0.036986  | -2.119131  |
| Yawing Moment wrt ADA (CMZA)   | Cny  | 0          | 0          |
| CG location, hcg=Xcg-Xref/Cref | hcg  | 0.249987   | 0.249987   |
| Neutral point, hn=hcg-Cma/CLa  | hn   | 0.668805   | 0.668805   |
| FX variation with ADY          | Cxb  | 0          | 0          |
| Side force derivative          | Cyb  | -0.006727  | -0.385454  |
| FZ variation with ADY          | Czb  | 0          | 0          |
| Dihedral Effect (CMXB)         | Clb  | -0.001824  | -0.104523  |
| Pitching Moment wrt ADY (CMYB) | Cmb  | 0          | 0          |
| Directional Stability (CMZB)   | Cnb  | 0.002995   | 0.171624   |

Figure 19: Stability derivatives for the model.

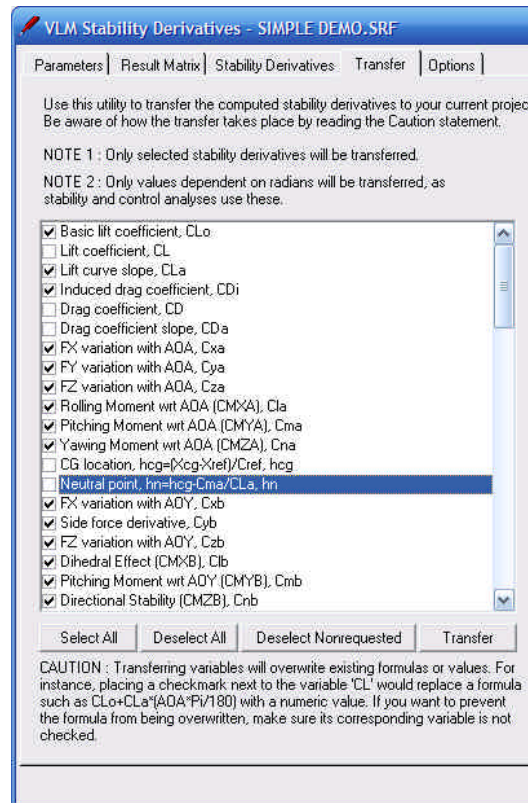
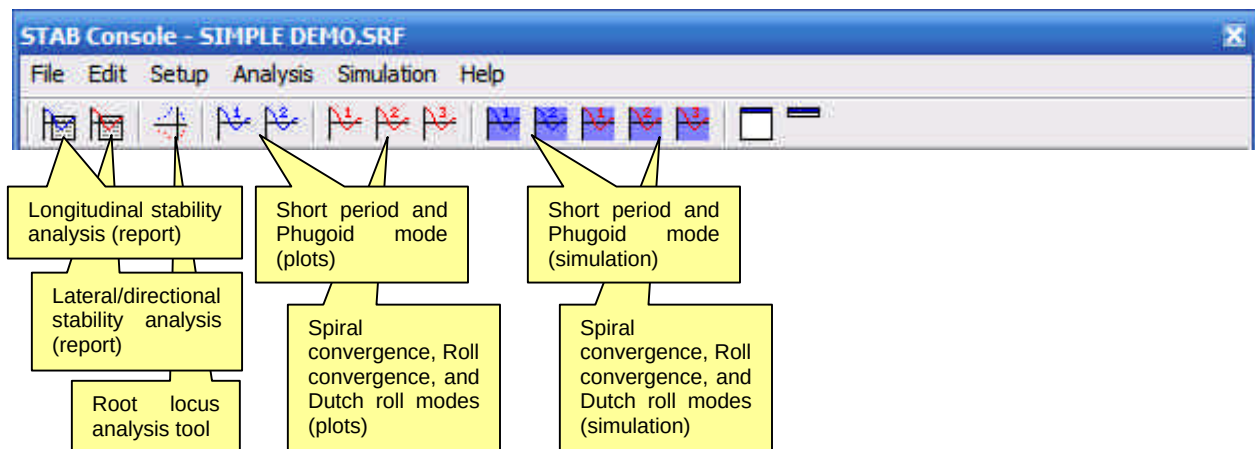


**Figure 20: Stability derivatives for the model.**

**STEP 28: Press the STAB Console icon. This will open the Stability Analysis Console shown in Figure 21.**



It is left as an exercise for the user to press the various icons to experience functionality. The simulation icons will display the motion of the aircraft in real time.



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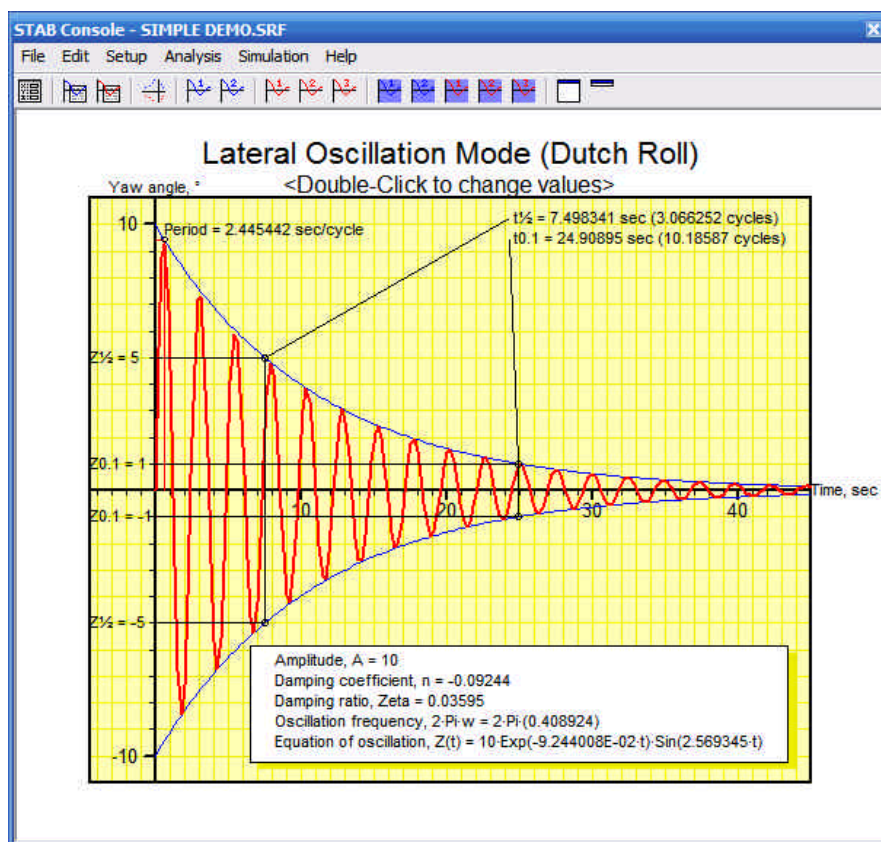
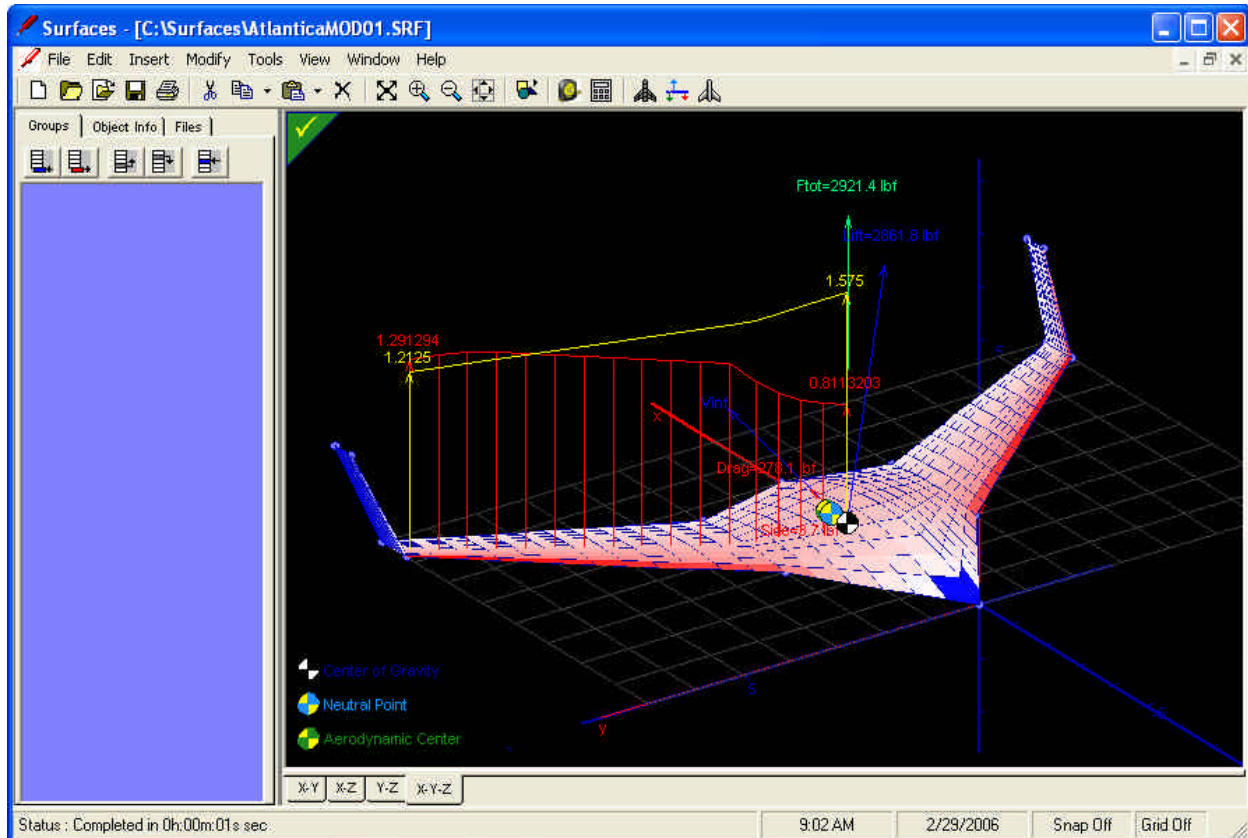


Figure 21: Stability analysis module.

## Accomplishing Special Projects with SURFACES

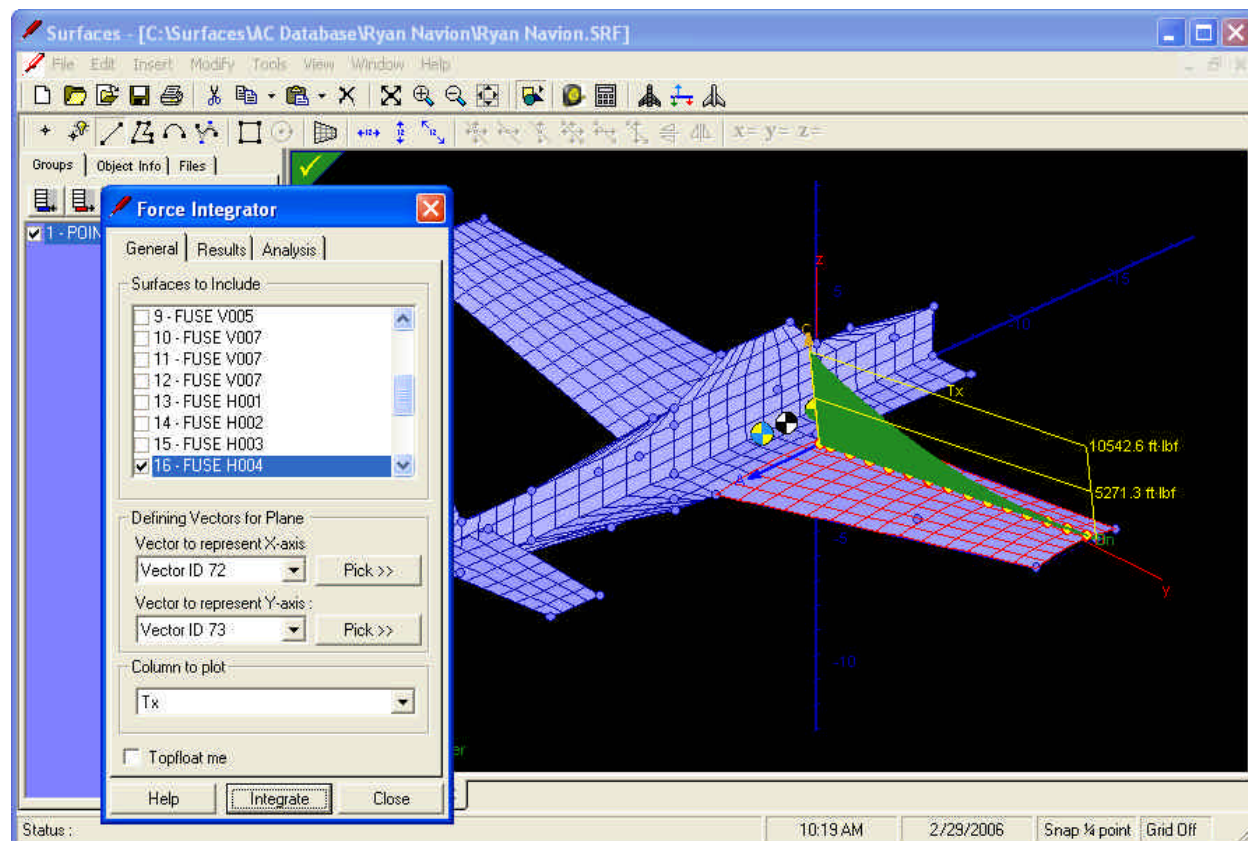
### *Tailoring Wings to Improve Stall Characteristics*



The above image shows how **SURFACES** can be used to help optimize stall characteristics. The yellow line represents section lift coefficients at stall. These are entered as reference values for curves A1 and A2 for each surface. The red lines represent section lift coefficients at the flight condition. The image shows the wing tip stalls long before the inboard part of the wing. Not only would this cause the airplane to a roll at stall (as one wing tip is prone to stalling before the other one), but more seriously, would result in an uncontrollable nose pitch-up moment. This situation can be remedied by modifying the wing geometry, for instance by adding wing washout, increase tip chord, reduce sweep, or using airfoils with a higher max lift coefficient.

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### *Determine Shear, Moment, and Torsion*



**SURFACES** comes equipped with a tool that allows you to analyze cantilevered shear and moment acting on any surface. Here shown is the Force Integrator tool as applied to the right wing on the Ryan Navion model. The bending moments along the right wing are plotted. Note the wing curvature represents the camberline of the aircraft's airfoils.

# Transformation of Load Vectors from a Global to a Local Coordinate System

The following derives mathematical formulation to determine shear forces and moments about an arbitrary axis. The goal is to provide **SURFACES** with a tool that helps the structural analyst retrieve aerodynamic loads. However, the formulation is in fact applicable to any load analysis involving a discrete distribution of elemental loads.

Consider a lifting surface in a 3D coordinate system (from now on referred to as the **global coordinate system**). For structural purposes it is desired to determine the shear and moments about an axis, called the *quarter chord*. **SURFACES** allows this to be done quickly and effectively. The analysis requires a coordinate system to be constructed, which from now on referred to as the **local coordinate system**.

A more descriptive example of this is shown with the typical Vortex-Lattice model in Figure 1. A vector on the leading edge and along the fuselage have been highlighted (in red). Additionally, the right wing has been highlighted. With this information, it is now possible to determine the 3D shear and moment distribution along either vector, due to the discrete elemental forces generated by the right wing. The two vectors are necessary to create the **local coordinate system** about which the shear and moments are resolved. Consequently, they are referred to as the *basis* of the local coordinate system. This way, one can analyze loads along vectors of arbitrary orientation.

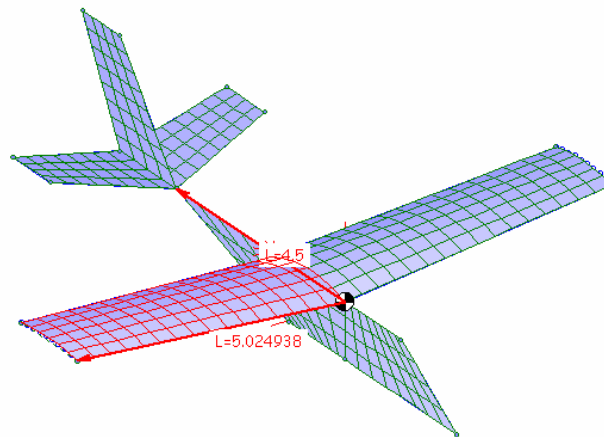


Figure 1: A typical Vortex-Lattice model.

## 1. Establishment of a Local Coordinate System

Consider the force  $F$  generated by an arbitrary panel in the global coordinate system  $X-Y-Z$  as shown in Figure 2.

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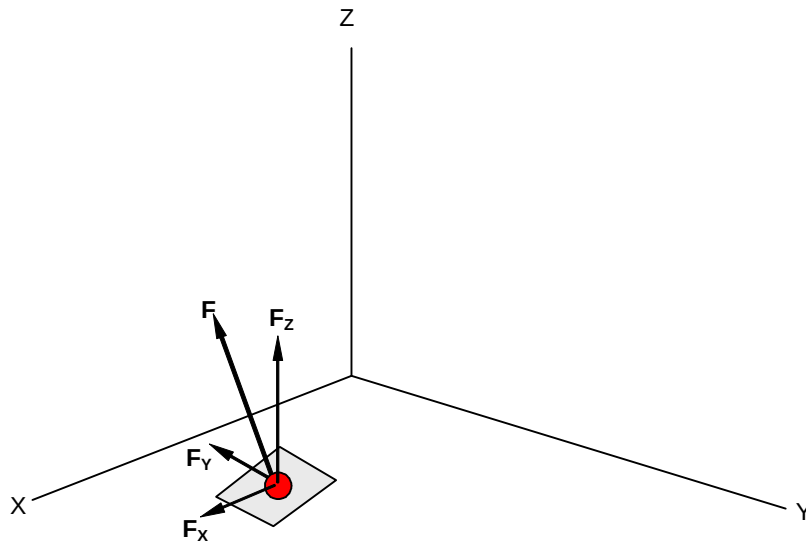


Figure 2: A force in the global coordinate system.

Consider a local coordinate system, identified by the selection of two vectors, **A** and **B**, such that **A** is not parallel to **B** (see Figure 3). These vectors uniquely define a plane (and are thus the basis of the coordinate system), whose normal is given by the vector **C**, such that:

$$\mathbf{C} = \mathbf{A} \times \mathbf{B} \quad (1)$$

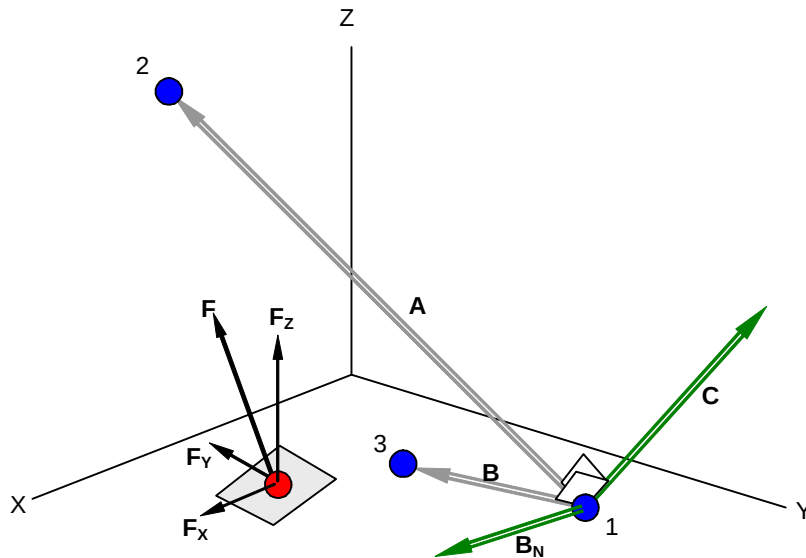


Figure 3: Defining the local coordinate system.

We can now create a local coordinate system, denoted by the vectors **A**, **B<sub>N</sub>**, and **C**, where **B<sub>N</sub>** is given by

$$\mathbf{B}_N = \mathbf{A} \times \mathbf{C} \quad (2)$$

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Note that the three vectors form a mutually perpendicular coordinate system. The determination of  $\mathbf{B}_N$  is necessary as  $\mathbf{B}$  may or may not be perpendicular to the vector  $\mathbf{A}$ .

Also note that according to convention, the vector  $\mathbf{A}$  represents the X-axis of the local coordinate system, here denoted by the lower case letters x-y-z. The vectors  $\mathbf{B}_N$  and  $\mathbf{C}$  correspond to the Y and Z axes, respectively.

Finally, note that the unit vectors for the local coordinate system are denoted as follows:

Unit vector for A:  $\begin{bmatrix} u_{AX} & u_{AY} & u_{AZ} \end{bmatrix}$

Unit vector for B:  $\begin{bmatrix} u_{BX} & u_{BY} & u_{BZ} \end{bmatrix}$

Unit vector for C:  $\begin{bmatrix} u_{CX} & u_{CY} & u_{CZ} \end{bmatrix}$

## 2. Transformation of Force Vector in Coordinate System A-B<sub>N</sub>-C

The force vector,  $\mathbf{F}$ , represented as  $\{F_x, F_y, F_z\}$  or  $F_x \cdot \mathbf{i} + F_y \cdot \mathbf{j} + F_z \cdot \mathbf{k}$  in the global coordinate system can now be represented as a force in the local one as  $\{F_x, F_y, F_z\}$  (see Figure 4). This is accomplished with a simple transformation of the vector  $\mathbf{F}$  onto the three vectors  $\mathbf{A}$ ,  $\mathbf{B}_N$ , and  $\mathbf{C}$  using the matrix notation of Equation (3).

$$\begin{Bmatrix} F_x \\ F_y \\ F_z \end{Bmatrix} = \begin{bmatrix} u_{AX} & u_{AY} & u_{AZ} \\ u_{BX} & u_{BY} & u_{BZ} \\ u_{CX} & u_{CY} & u_{CZ} \end{bmatrix} \begin{Bmatrix} F_x \\ F_y \\ F_z \end{Bmatrix} \quad (3)$$

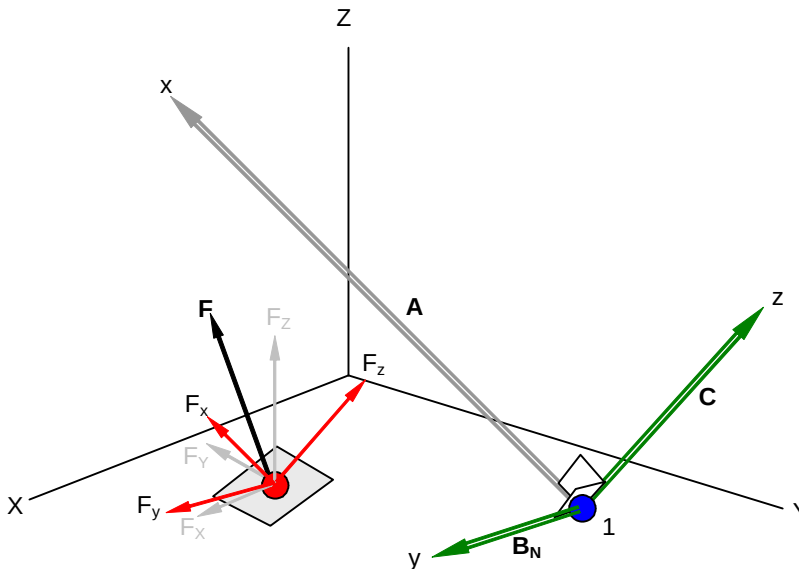


Figure 4: Transformation of vector  $\mathbf{F}$ .

### Example:

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The force vector  $\mathbf{F} = F_x \cdot \mathbf{i} + F_y \cdot \mathbf{j} + F_z \cdot \mathbf{k} = 10 \cdot \mathbf{i} - 5 \cdot \mathbf{j} + 10 \cdot \mathbf{k}$  is given in a global coordinate system. Two vectors  $\mathbf{A}$  and  $\mathbf{B}$  are given as the basis for our local coordinate system as follows:

$$\mathbf{A} = -\mathbf{i} - \mathbf{j} + \mathbf{k}$$

$$\mathbf{B} = 0.5\mathbf{i} - \mathbf{j} + 0.5\mathbf{k}$$

Determine the components of  $\mathbf{F}$  in the local coordinate system created by the vectors  $\mathbf{A}$  and  $\mathbf{B}$ .

**Solution:**

Step 1: Determine the vector  $\mathbf{C}$  from  $\mathbf{C} = \mathbf{A} \times \mathbf{B}$ .

$$\mathbf{C} = \mathbf{A} \times \mathbf{B} = \begin{vmatrix} \mathbf{i} & \mathbf{j} & \mathbf{k} \\ -1 & -1 & 1 \\ 0.5 & -1 & 0.5 \end{vmatrix} = 0.5\mathbf{i} + \mathbf{j} + 1.5\mathbf{k}$$

Step 2: Determine the vector  $\mathbf{B}_N$  from  $\mathbf{B}_N = \mathbf{A} \times \mathbf{C}$ .

$$\mathbf{B}_N = \mathbf{A} \times \mathbf{C} = \begin{vmatrix} \mathbf{i} & \mathbf{j} & \mathbf{k} \\ -1 & -1 & 1 \\ 0.5 & 1 & 1.5 \end{vmatrix} = -2.5\mathbf{i} + 2\mathbf{j} - 0.5\mathbf{k}$$

Step 3: Determine force component per Equation (5). Start by determining the unit vectors and assemble into the transformation matrix:

$$\begin{bmatrix} u_{AX} & u_{AY} & u_{AZ} \\ u_{BX} & u_{BY} & u_{BZ} \\ u_{CX} & u_{CY} & u_{CZ} \end{bmatrix} = \begin{bmatrix} -0.57735 & -0.57735 & 0.57735 \\ -0.77152 & 0.61721 & -0.15430 \\ 0.26726 & 0.53452 & 0.80178 \end{bmatrix}$$

This yields the following force components using Equation (3):

$$\begin{Bmatrix} F_x \\ F_y \\ F_z \end{Bmatrix} = \begin{bmatrix} -0.57735 & -0.57735 & 0.57735 \\ -0.77152 & 0.61721 & -0.15430 \\ 0.26726 & 0.53452 & 0.80178 \end{bmatrix} \begin{Bmatrix} 10 \\ -5 \\ 10 \end{Bmatrix} = \begin{Bmatrix} 2.8868 \\ -12.3443 \\ 8.0178 \end{Bmatrix}$$

### 3. Determination of Moment Vector in Coordinate System A-B<sub>N</sub>-C

As stated in the introduction, ultimately, the goal of the analysis presented herein is the determination of shear forces and moments about an axis due to the cumulative effects of multiple discrete forces. It was demonstrated in Section 2 how shear forces are transformed to a local coordinate system. The same methodology can be applied to the generation of moments, but it involves a transformation about a point, P, through which the vector  $\mathbf{A}$  goes. This point will be called the *projection point* from now on. It is the projection of the point  $(x_F, y_F, z_F)$  on to the vector  $\mathbf{A}$  (see Figure 6). It is denoted by the point  $(x_P, y_P, z_P)$ .

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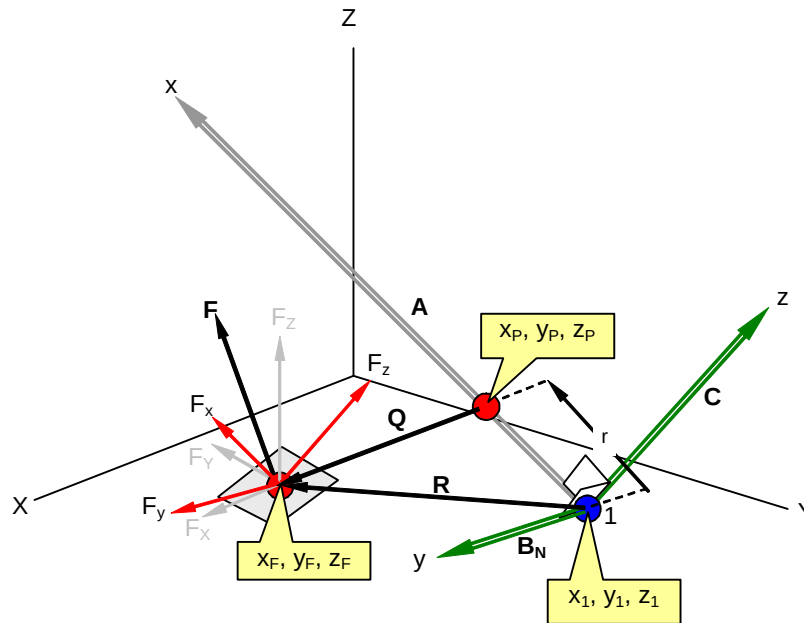


Figure 6: Determination of moment vector  $\mathbf{M}$ .

The location of this point is obtained using standard vector algebra. The reader is referred to the one presented on page 31 in *Introduction to Vector Analysis*, by Davis and Snyder. The method can be explained using Figure 7, which defines the arbitrary vectors  $\mathbf{V}$  and  $\mathbf{W}$ .

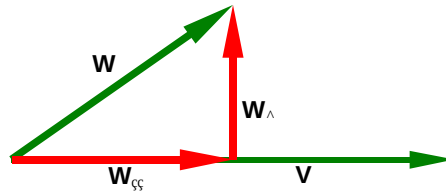


Figure 7: Projection of vector  $\mathbf{W}$  onto vector  $\mathbf{V}$ .

Then, the parallel projection of  $\mathbf{W}$  onto  $\mathbf{V}$  is given by:

$$\mathbf{W}_{\parallel} = \left( \frac{\mathbf{V} \cdot \mathbf{W}}{\mathbf{V} \cdot \mathbf{V}} \right) \mathbf{V} \quad (4)$$

The perpendicular projection is simply found from:

$$\mathbf{W}_{\perp} = \mathbf{W} - \mathbf{W}_{\parallel} \quad (5)$$

Using this, we first determine the vector  $\mathbf{R}$  from the start point of the vector  $\mathbf{A}$  to the force point, i.e.:

$$\mathbf{R} = \begin{Bmatrix} x_F - x_1 \\ y_F - y_1 \\ z_F - z_1 \end{Bmatrix} \quad (6)$$

The location is then found by referencing Figure 7 and Equation (4) and by writing:

$$\begin{Bmatrix} x_p \\ y_p \\ z_p \end{Bmatrix} = \begin{Bmatrix} x_1 \\ y_1 \\ z_1 \end{Bmatrix} + \left( \frac{\mathbf{A} \cdot \mathbf{R}}{\mathbf{A} \cdot \mathbf{A}} \right) \mathbf{A} \quad (7)$$

The length of the parallel projection (the rightmost term of Equation (7)) is denoted by the letter  $r$ . It will be used in Section 4 to sort the discrete loads and moments along the vector  $\mathbf{A}$ . Now, one must determine the vector from the projection point to the force point, denoted by  $\mathbf{Q}$ . This vector is given by Equation (8):

$$\mathbf{Q} = \begin{Bmatrix} x_F - x_p \\ y_F - y_p \\ z_F - z_p \end{Bmatrix} \quad (8)$$

Then, calculate the discrete moment about the projection point from

$$\mathbf{M} = \mathbf{F} \times \mathbf{Q} = \begin{vmatrix} i & j & k \\ F_x & F_y & F_z \\ Q_x & Q_y & Q_z \end{vmatrix} \quad (9)$$

The moment vector,  $\mathbf{M}$ , represented as  $\{M_x, M_y, M_z\}$  is still in the global coordinate system. It can now be treated as the force in the local one, i.e. as  $\{\mathbf{M}_x, \mathbf{M}_y, \mathbf{M}_z\}$  using the same transformation as for the force vector.

$$\begin{Bmatrix} M_x \\ M_y \\ M_z \end{Bmatrix} = \begin{bmatrix} u_{AX} & u_{AY} & u_{AZ} \\ u_{BX} & u_{BY} & u_{BZ} \\ u_{CX} & u_{CY} & u_{CZ} \end{bmatrix} \begin{Bmatrix} M_x \\ M_y \\ M_z \end{Bmatrix} \quad (10)$$

#### 4. Determination of Shear and Moment Distribution

Figure 7 shows several loads whose components have been transformed to the local coordinate system specified by  $\mathbf{A}$ ,  $\mathbf{B}_N$ , and  $\mathbf{C}$ . Each has associated force and moment components and the parameter  $r$ , which is simply the distance of the projection point from the starting point of vector  $\mathbf{A}$  (point 1). The purpose of the parameter  $r$  is to allow sorting to take place (say from start towards the end of the vector  $\mathbf{A}$ ). The sorted components are then used to construct shear and moment diagrams in a standard fashion.

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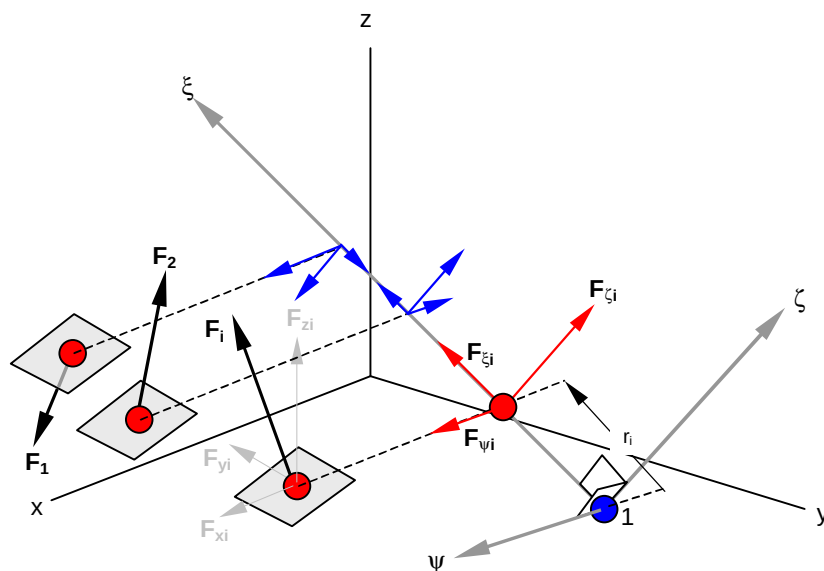


Figure 7: Methodology for construction shear and moment diagrams.

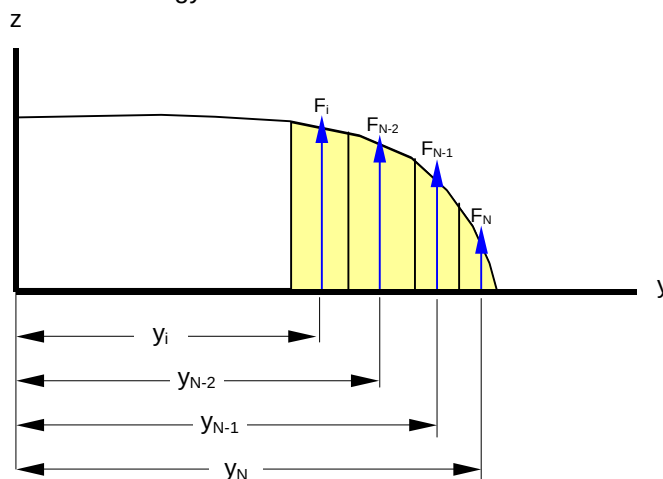


Figure 8: Creation of shear and moment diagrams from discrete forces.

#### 4.1 Approximation for Shear in the Z-direction Along the Y-axis Vector

Approximating shear forces is simple, just apply Equation (11),

$$V_{zi} \approx \sum_{i=j}^N F_{zi} \quad (11)$$

#### 4.2 Approximation for Moment about X-axis Along the Y-axis Vector

The approximation for the moments is implemented as follows. The moment at  $N-1$  is due to the force  $F_{zN}$  acting at a distance  $y_N - y_{N-1}$ . Similarly, the moment at  $N-2$  is due to the force  $F_{zN}$  acting at a distance  $y_N - y_{N-2}$  and the force  $F_{zN-1}$  acting at a distance  $y_{N-1} - y_{N-2}$ . Writing this in a general form leads to:

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$$\begin{aligned}
 M_{xi} &\approx F_{zi+1} \cdot (y_{i+1} - y_i) + \dots + F_{zN-2} \cdot (y_{N-2} - y_i) + F_{zN-1} \cdot (y_{N-1} - y_i) + F_{zN} \cdot (y_N - y_i) \\
 &= \sum_{j=i+1}^N F_{zj} \cdot (y_j - y_i)
 \end{aligned}
 \tag{12}$$

#### 4.3 Approximation for Torsion About the Y-axis Vector

The approximation for the torsion is implemented as follows. The torsion at  $N-1$  is due to the force  $F_N$  acting at an offset distance of  $x_N - x_{pN}$ , where  $x_p$  denotes the  $x$  value of the projection point. Similarly, the moment at point  $N-1$  is due to the force  $F_{zN}$  acting at a distance  $x_N - x_{pN}$  and the force  $F_{zN-1}$  acting at a distance  $x_{N-1} - x_{pN-1}$ . Writing this in a general form leads to:

$$\begin{aligned}
 M_{yi} &\approx F_{zi} \cdot (x_i - x_{pi}) + \dots + F_{zN-2} \cdot (x_{N-2} - x_{pN-2}) + F_{zN-1} \cdot (x_{N-1} - x_{pN-1}) + F_{zN} \cdot (x_N - x_{pN}) \\
 &= \sum_{j=i}^N F_{zj} \cdot (x_j - x_{pj})
 \end{aligned}
 \tag{13}$$

#### Example:

A lifting surface is 10 ft long (span) and 2 ft wide (chord). It carries a uniform pressure load of 1 lb<sub>f</sub>/ft<sup>2</sup>. Determine the shear in the  $z$ -direction, moment about the  $x$ -axis, and torsion about the  $y$ -axis at  $y=0.5$  ft, assuming the span to be partitioned into 10, 1 ft wide strips. Note that each strip will carry 2 lb<sub>f</sub> of load.

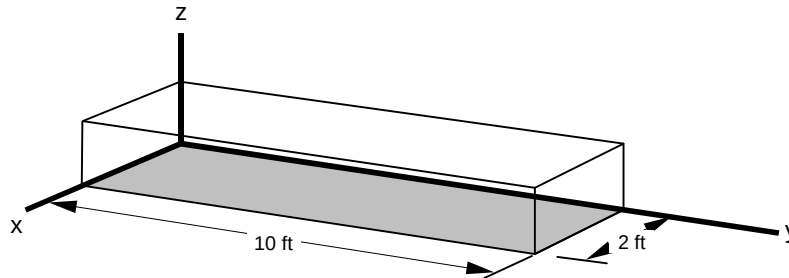


Figure 9: Lifting surface with a uniform pressure distribution.

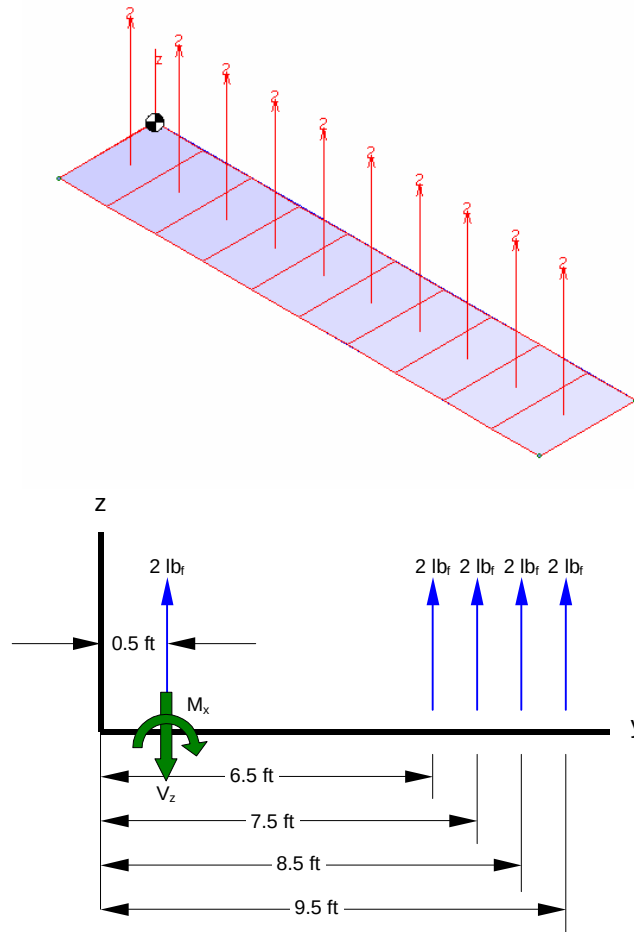


Figure 10: Discrete forces replace the uniform distribution. Reaction forces are shown in green.

**Solution:**

Shear is determined from Equation (11):

$$V_i \approx \sum_{i=1}^N F_i = 2 + 2 + \dots + 2 = 20 \quad lb_f$$

Moment is determined from Equation (12):

$$\begin{aligned} M_x &\approx F_2 \cdot (y_2 - y_1) + \dots + F_8 \cdot (y_8 - y_1) + F_9 \cdot (y_9 - y_1) + F_{10} \cdot (y_{10} - y_1) \\ &= 2 \cdot (1) + \dots + 2 \cdot (7) + 2 \cdot (8) + 2 \cdot (9) \\ &= 90 \quad ft \cdot lb_f \end{aligned}$$

Torsion is determined from Equation (13):

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$$\begin{aligned}
 M_y &\approx F_1 \cdot (x_1 - x_{p1}) + \dots + F_8 \cdot (x_8 - x_{p8}) + F_9 \cdot (x_9 - x_{p9}) + F_{10} \cdot (x_{10} - x_{p10}) \\
 &= 2 \cdot (1) + \dots + 2 \cdot (1) + 2 \cdot (1) + 2 \cdot (1) \\
 &= 20 \text{ ft} \cdot \text{lb}_f
 \end{aligned}$$

The exact value for the shear is determined from  $V = w \cdot A = (1 \text{ lb}_f/\text{ft}^2) \cdot (10 \text{ ft} \times 2 \text{ ft}) = 20 \text{ lb}_f$ .

Similarly (noticing that the centroid of the force  $V$  is at  $y=5 \text{ ft}$ ), the moment about a point  $y = 0.5 \text{ ft}$  (necessitated by the discreteness of the strip solution) is  $M_x = V \cdot \Delta y = (20 \text{ lb}_f) \cdot (5 \text{ ft} - 0.5 \text{ ft}) = 90 \text{ ft} \cdot \text{lb}_f$ .

Finally, noticing the the centroid of the force  $V$  is at  $x=1 \text{ ft}$ , we find that  $M_y = V \cdot \Delta x = (20 \text{ lb}_f) \cdot (1 \text{ ft}) = 20 \text{ ft} \cdot \text{lb}_f$ .

## 5. Presentation of Data in SURFACES

The user selects **Results -> Force Integrator...** from the VLM Console in **SURFACES** as shown in Figure 11 below.

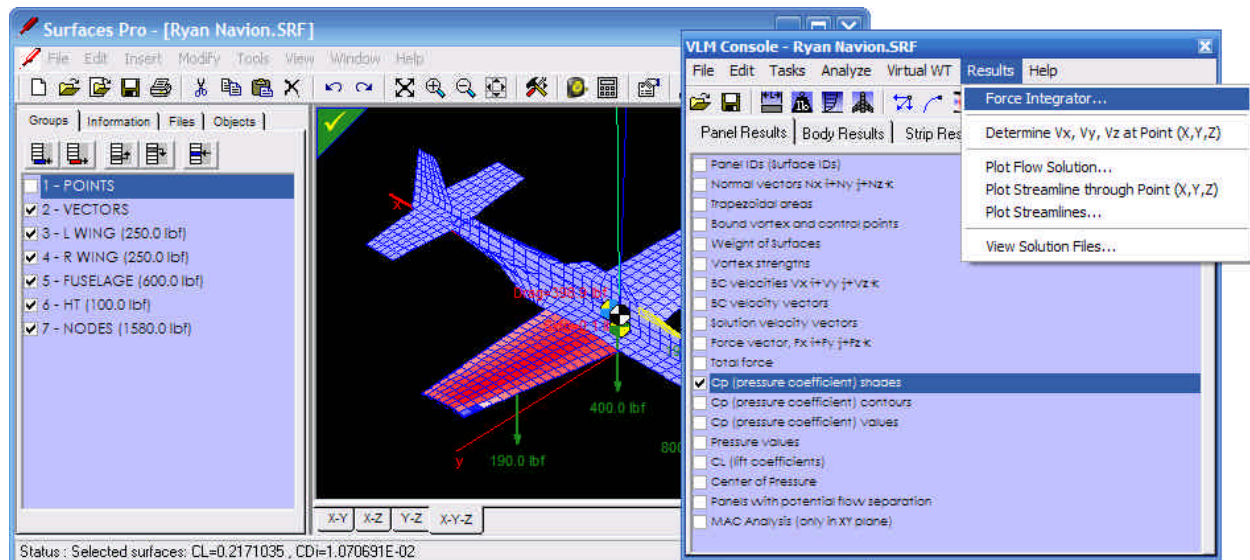


**Figure 11: Selecting the Force Integrator tool.**

Once the pertinent surfaces and vectors (corresponding to vectors **A** and **B**) have been selected, the user can press the *Integrate* button as shown in Figure 12. Selecting the Results tab will display a table with analysis results. Table 2 details the heading names.

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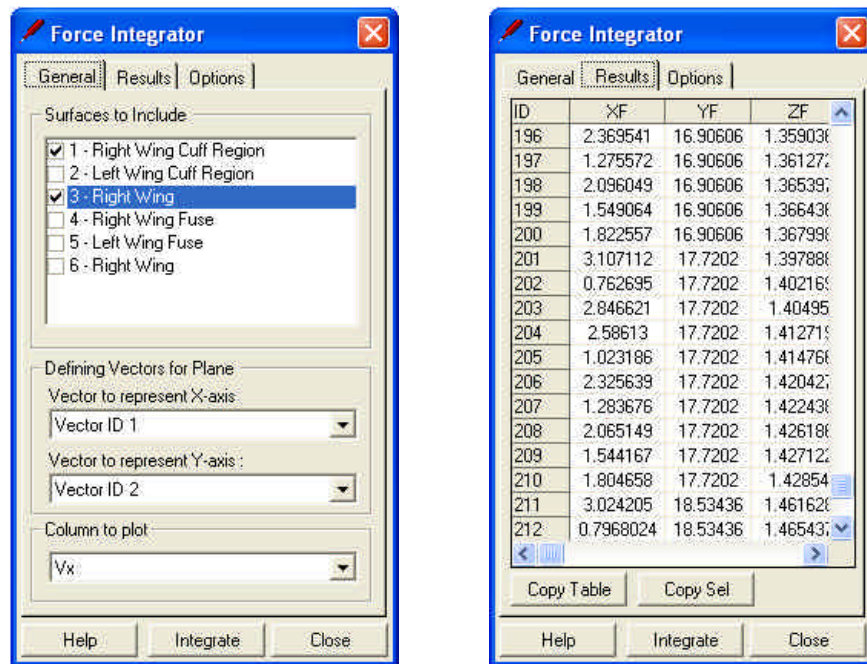


Figure 12: Force Integrator tool.

Table 2: Heading Names

| Heading    | Description                                                                      |
|------------|----------------------------------------------------------------------------------|
| XF, YF, ZF | X, Y, and Z-coordinates of the panel force, which is its centroid.               |
| XP, YP, ZP | X, Y, and Z-coordinates of the panel force panel force projection onto vector A. |
| r          | Distance from Point 1 of vector A to XP,YP,ZP.                                   |
| Rx, Ry, Rz | Components of the vector R, from Point 1 of vector A.                            |
| R          | The length of vector R.                                                          |
| Qx, Qy, Qz | Components of the vector Q, from XP, YP, ZP to XF, YF, ZF.                       |
| Q          | The length of vector Q.                                                          |

'Panel force (body system) in global coordinate system

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Fbx"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Fby"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Fbz"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "|Fb|"

'Panel force (airspeed system) in global coordinate system

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Fx"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Fy"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Fz"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "|F|"

'Panel moment in global coordinate system

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Mx"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "My"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Mz"

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "|M|"

'Panel force in global coordinate system

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```

i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Ftx"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Fty"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Ftz"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "|Ft|"
'Panel moment in global coordinate system
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Mtx"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Mty"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Mtz"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "|Mt|"
'Panel force in local coordinate system
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Vx"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Vy"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Vz"
'Panel moment in local coordinate system
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Tx"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Ty"
i = i + 1: gridCntrl.Col = i: gridCntrl.Text = "Tz"

```

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## Using the Virtual Wind Tunnel

The Virtual Wind Tunnel (VWT) allows you to analyze your model exactly as if you were to run it in a real wind tunnel. You can vary several parameters from an initial value to a final value in prescribed steps. For instance, you can perform an *alpha*- or a *beta*-sweep, exactly as you would do it in a real tunnel, but without the hassle. Before you use the tunnel, you must understand its limitations.

Any good airplane design operates most of its lifetime at airspeeds at which the airflow is relatively smooth and at a low angles-of-attack (AOA) and yaw (AOY). The lifting surfaces are always sized such that this is achievable. The primary advantage is that drag is minimum at such conditions and, therefore, the airplane is the most efficient. Under these circumstances, forces and moments change linearly with these angles. However, when the airplane slows down before it lands, or for some other extreme maneuvering, it begins to operate at larger AOAs and AOYs, causing the flow to separate. This will introduce a nonlinearity into forces and moments. Linear codes, including **SURFACES**, do not account for this phenomena.

At this point you may be asking yourself, why then resort to linear analysis if it has this shortcoming? The answer is as simple as it is resounding. Speed! Accuracy is an additional benefit if your model is well created. But the primary reason is speed. Linear analysis is extremely fast when compared to nonlinear analysis. At the time of this writing, using **SURFACES** one can create and analyze an aircraft in the linear range with an incredible accuracy in a matter of minutes. The same model may take 4-6 weeks to prepare for a nonlinear Navier-Stokes solver, and would give one (yes one) AOA, say every 24 hours, if one's computer network holds up. And, you should ask the question; But isn't the Navier-Stokes (N-S) method more accurate? The answer is yes and no. In fact, in the linear range, it will give a similar answer as the Vortex-Lattice Method (VLM), it will just take much, much longer to get those answers. The person writing these words has experienced many times that the VLM has been closer to actual wind tunnel data than N-S. The strength of N-S solvers is separated flow, but at this time, such tools are better at giving the aerodynamicist an idea of what the flow field looks like than trustworthy coefficients.

Naturally, it must be emphasized that **SURFACES** is performing a mathematical simulation when you use its wind tunnel test tool. The same rule applied to all computer codes that emulate wind tunnels; a real wind tunnel test always overrides any such calculations (assuming the data was obtained by reliable means). However, assuming you are using **SURFACES** to create a mathematical model of your design, the VWT is a great tool to help you understand the following issues:

- (1) The AOA and AOY, the airspeeds, and the rotation rates (P, Q, R) where your math model breaks down. You will want to know at which AOA the linear assumption breaks down.
- (2) Features of your model that, well, still need to be improved before an accurate comparison can be made of existing wind tunnel. The concept of tuning is well known in the world of finite elements, flutter, and linear modeling and the like. Tuning is done by making minor changes to the model un

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## Determination of a Trimmed Flight Condition

The following derivations details the requirements for a trimmed flight condition. A trimmed flight is defined as a flight in which the moments about all three axes is zero. For instance, when determining longitudinal trim (assuming a solution can be found) the following must hold:

$$\begin{aligned} C_L &= C_{L0} + C_{L\alpha} \cdot \alpha + C_{L\delta_e} \cdot \delta_e \\ C_m &= C_{m0} + C_{m\alpha} \cdot \alpha + C_{m\delta_e} \cdot \delta_e \end{aligned} \quad (1)$$

Where;  $C_L$  = lift coefficient

$C_{L0}$  = lift coefficient for zero angle-of-attack and zero elevator deflection

$C_{L0\delta_e}$  = lift coefficient for zero angle-of-attack

$C_{Ltarget}$  = lift coefficient obtained from the lift equation

$C_{L\alpha}$  = lift coefficient change with angle-of-attack

$C_{L\delta_e}$  = lift coefficient change with elevator deflection

$C_m$  = pitching moment coefficient

$C_{m0}$  = pitching moment coefficient for zero angle-of-attack and zero elevator deflection

$C_{m0\delta_e}$  = pitching coefficient for zero angle-of-attack

$C_{m\alpha}$  = pitching moment change with angle-of-attack

$C_{m\delta_e}$  = pitching moment change with elevator deflection

$\alpha$  = angle-of-attack

$\alpha_{CURR}$  = current angle-of-attack

$\Delta\alpha$  = deviation from current angle-of-attack

$\delta_e$  = elevator deflection

$\Delta\delta_e$  = deviation elevator deflection

If the coefficient are known, we can write Equation (1) as follows:

$$\begin{aligned} C_{L\alpha} \cdot \alpha + C_{L\delta_e} \cdot \delta_e &= C_L - C_{L0} \\ C_{m\alpha} \cdot \alpha + C_{m\delta_e} \cdot \delta_e &= C_m - C_{m0} \end{aligned} \Leftrightarrow \begin{bmatrix} C_{L\alpha} & C_{L\delta_e} \\ C_{m\alpha} & C_{m\delta_e} \end{bmatrix} \begin{Bmatrix} \alpha \\ \delta_e \end{Bmatrix} = \begin{Bmatrix} C_L - C_{L0} \\ C_m - C_{m0} \end{Bmatrix} \quad (2)$$

The solution protocol is a follows:

**STEP 1:** Compute:  $C_{Ltarget} = \frac{2W}{\rho V^2 S}$

**STEP 2:** Establish a value for  $\Delta\alpha$  and  $\Delta\delta_e$ . Set  $\alpha = \alpha_{CURR} - \Delta\alpha$  and  $\delta_e = 0$ , to determine  $C_{L1}$  and  $C_{m1}$ .

**STEP 3:** Set  $\alpha = \alpha_{CURR} + \Delta\alpha$  and  $\delta_e = 0$ , to determine  $C_{L2}$  and  $C_{m2}$ .

**STEP 4:** Compute  $C_{L\alpha}$  and  $C_{m\alpha}$  from:

$$\begin{cases} C_{L1} = C_{L0} + C_{L\alpha} \cdot \alpha_1 \\ C_{L2} = C_{L0} + C_{L\alpha} \cdot \alpha_2 \end{cases} \Rightarrow C_{L\alpha} = \frac{C_{L2} - C_{L1}}{\alpha_2 - \alpha_1} \quad \text{and} \quad C_{L0} = C_{L2} - C_{L\alpha} \cdot \alpha_2$$

and 
$$\left. \begin{aligned} C_{m1} &= C_{m0} + C_{m\alpha} \cdot \alpha_1 \\ C_{m2} &= C_{m0} + C_{m\alpha} \cdot \alpha_2 \end{aligned} \right\} \Rightarrow C_{m\alpha} = \frac{C_{m2} - C_{m1}}{\alpha_2 - \alpha_1} \quad \text{and} \quad C_{m0} = C_{m2} - C_{m\alpha} \cdot \alpha_2$$

**STEP 5:** Compute: 
$$\alpha_{\text{target}} = \frac{C_{L\text{target}} - C_{L0}}{C_{L\alpha}}$$

**STEP 6:** Set  $\alpha = \alpha_{\text{target}}$  and  $\delta_e = \delta_{e\text{CURR}} - \Delta\delta_e$ , to determine  $C_{L3}$  and  $C_{m3}$ .

**STEP 7:** Set  $\alpha = \alpha_{\text{target}}$  and  $\delta_e = \delta_{e\text{CURR}} + \Delta\delta_e$ , to determine  $C_{L4}$  and  $C_{m4}$ .

**STEP 8:** Compute  $C_{L\delta_e}$  and  $C_{m\delta_e}$  from: 
$$C_{L\delta_e} = \frac{C_{L3} - C_{L0}}{\delta_e} \quad \text{and} \quad C_{m\delta_e} = \frac{C_{m3} - C_{m0}}{\delta_e}$$

**STEP 9:** Compute the required  $C_L$  to support the desired lift and knowing that  $C_m=0$  for a balanced condition we populate the matrix of Equation (2) as follows:

$$\begin{bmatrix} C_{L\alpha} & C_{L\delta_e} \\ C_{m\alpha} & C_{m\delta_e} \end{bmatrix} \begin{bmatrix} \alpha \\ \delta_e \end{bmatrix} = \begin{bmatrix} C_L - C_{L0} \\ -C_{m0} \end{bmatrix} \quad (3)$$

And solve for the  $\alpha$  and  $\delta_e$ , which define the trimmed condition.

**SURFACES** solves this using an iterative algorithm and can do so about each of the airplane's three axes. This is necessary because the deflection of a control surface modifies the geometry which, in turn, requires a new flow solution. The program comes with an easy to use Trim Wizard that makes this a breeze. Additionally, you can trim for multiple airspeeds, creating an individual flow solution for each airspeed. This is handy when you want multiple solutions for the same CG location. You can leave your computer overnight running the trim solutions, and study the solution files the next day.

## Validation Samples

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## Validation 1 - 2-D Flat Plate Airfoil

### V1.1 Model

A high aspect ratio (AR=20) wing model was constructed to obtain 2-D pressure coefficients for comparison to theoretical data. The model has a wing span of 20 units and a chord of 1 unit. The Angle-of-Attack is  $10^\circ$  at an airspeed of 10 unit/sec and density of 1 mass unit/length<sup>3</sup>. The  $C_p$  at the center of the model was obtained for 2, 5, 10, and 15 chord wise panel density. Each of the two surfaces has 34 span wise panels.

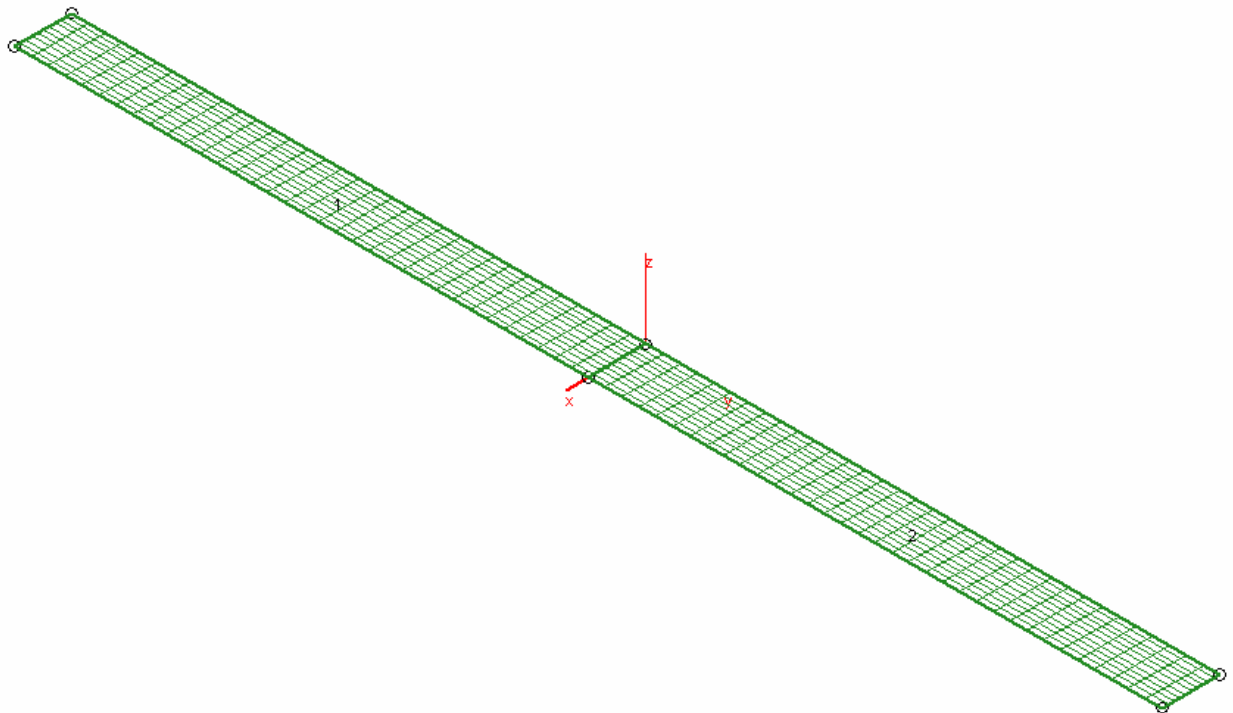


Figure 1-1: High aspect ratio wing used to evaluate the 2-D  $C_p$ .

### V1.2 Expected Result

Is obtained from the book *Aerodynamics, Aeronautics, and Flight Mechanics*, by Barnes W. McCormick. The data is obtained from Figure 3.17 on page 87.

### V1.3 Results from SURFACES

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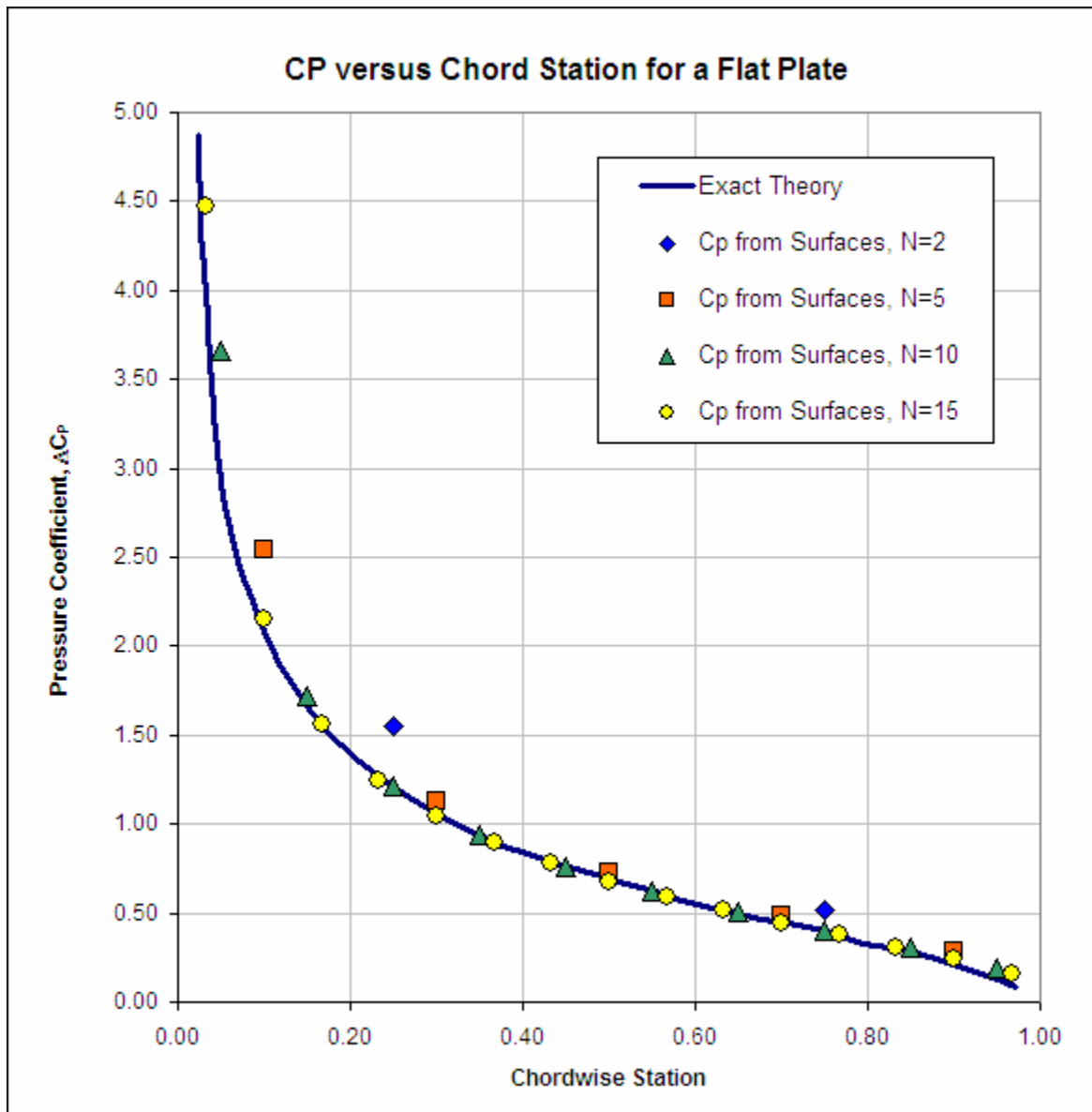
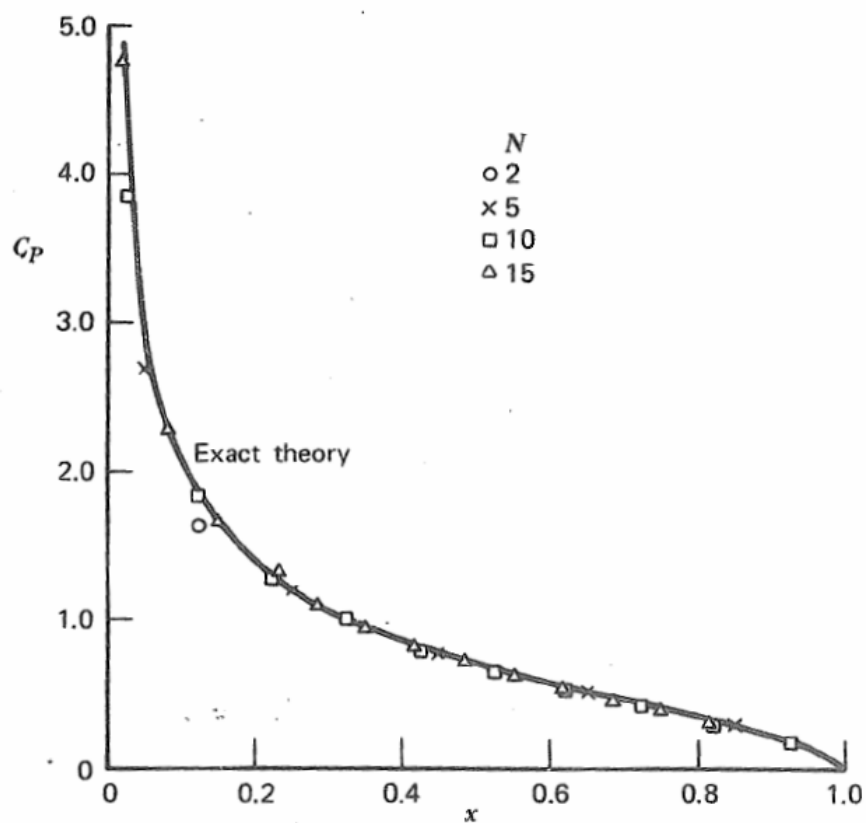


Figure 1-2: 2-D Cp for various panel densities from *SURFACES* compared to exact theory.

Data from Figure 3.17

| X    | CP   |
|------|------|
| 0.02 | 4.88 |
| 0.03 | 4.47 |
| 0.03 | 4.22 |
| 0.03 | 4.01 |
| 0.04 | 3.45 |
| 0.04 | 3.23 |
| 0.05 | 2.93 |
| 0.07 | 2.52 |
| 0.10 | 2.06 |
| 0.13 | 1.77 |
| 0.17 | 1.54 |
| 0.21 | 1.35 |
| 0.26 | 1.16 |
| 0.34 | 0.96 |
| 0.39 | 0.85 |
| 0.46 | 0.75 |
| 0.54 | 0.64 |
| 0.60 | 0.55 |
| 0.68 | 0.47 |
| 0.74 | 0.41 |
| 0.79 | 0.33 |
| 0.86 | 0.27 |
| 0.92 | 0.18 |
| 0.97 | 0.08 |



## Validation 2 - 3-D Properties of Two Wings

### V2.1 Models

Two moderately high aspect ratio wing models were constructed to compare results from the VLM to a standard 3-D aerodynamic analysis. The models have a wing span of 10 ft and a chord of 1 ft. One model has a 0° leading edge sweep and the other 35°. The angle of attack is 10° at an airspeed of 100 KCAS (168.8 ft/s) and density of 0.002378 slugs/ft<sup>3</sup>. Each of the two surfaces has 32 spanwise and 8 chord-wise panels.

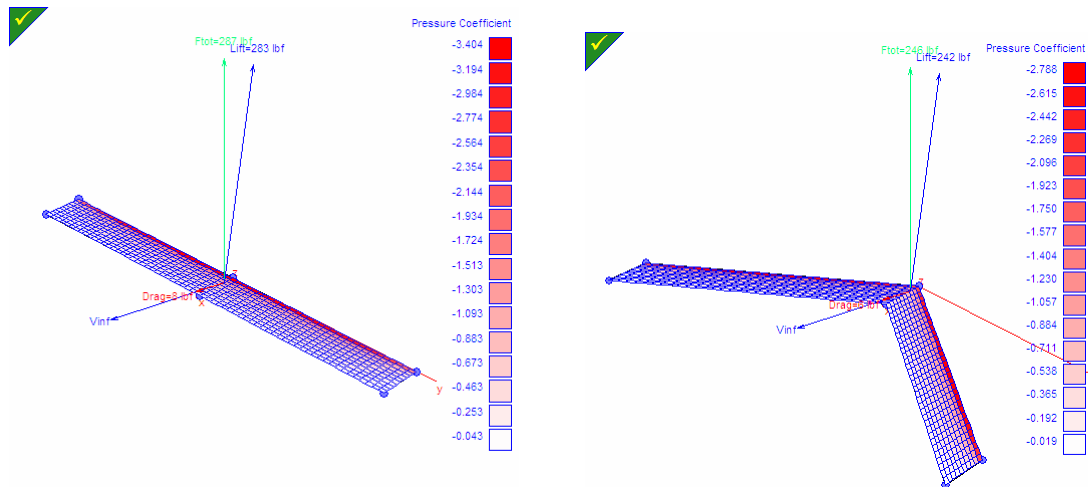


Figure 2-1: The two 3-D wing models.

### V2.2 Expected Result

The following parameters are given:

Airspeed  $V = 168.8$  ft/s ( $M = 168.8/1116 = 0.151$ )  
Wing area  $S = 10 \times 1 = 10$  ft<sup>2</sup>  
Aspect Ratio  $AR = b^2 / S = 10^2/10 = 10$

Assume a 2-D lift curve slope of  $C_{L_{\alpha 2D}} = 0.1063$  per deg (for NACA0009, from *Theory of Wing Sections*, by Abbott and Doenhoff). Start by computing a 3-D lift curve slope from Method 1 of USAF DATCOM Section 1, page 1-7.

$$C_{L_{\alpha}} = \frac{2\pi \cdot AR}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{C/2}}{\beta^2} \right)} + 4}$$

Where;  $AR$  = Wing Aspect Ratio = 10

$\beta$  = Mach number parameter (Prandtl-Glauert) =  $(1-M^2)^{0.5} = 0.989$

$\kappa$  = Ratio of 2D lift curve slope to  $2\pi = 0.1063 \times (180/\pi)/(2\pi) = 0.96934$

$\Lambda_{C/2}$  = Sweepback of mid-chord = 0° and 35°

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| $\Lambda_{c/2} = 0^\circ$                                                                                                                                                                                                                                                                                          | $\Lambda_{c/2} = 35^\circ$                                                                                                                                                                                                                                                                                                        |
|--------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|-----------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| $C_{L_\alpha} = \frac{2\pi \cdot AR}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{c/2}}{\beta^2} \right)} + 4}$ $= \frac{2\pi \cdot 10}{2 + \sqrt{\frac{100 \times 0.989^2}{0.96934^2} \left( 1 + \frac{0}{0.989^2} \right)} + 4}$ $= 5.068 \text{ per rad} = 0.08846 \text{ per deg}$ | $C_{L_\alpha} = \frac{2\pi \cdot AR}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{c/2}}{\beta^2} \right)} + 4}$ $= \frac{2\pi \cdot 10}{2 + \sqrt{\frac{100 \times 0.989^2}{0.96934^2} \left( 1 + \frac{\tan^2(35^\circ)}{0.989^2} \right)} + 4}$ $= 4.286 \text{ per rad} = 0.07480 \text{ per deg}$ |
| <p>The lift coefficient at <math>10^\circ</math> is thus:</p> $C_L = 10^\circ \times C_{L_\alpha} = 0.8846$                                                                                                                                                                                                        | <p>The lift coefficient at <math>10^\circ</math> is thus:</p> $C_L = 10^\circ \times C_{L_\alpha} = 0.7480$                                                                                                                                                                                                                       |
| <p>The total lift of the wing is</p> $L = \frac{1}{2} \rho V^2 S C_L$ $= \frac{1}{2} (0.002378) (168.8)^2 (10) (0.8846)$ $= 299.7 \text{ lb}_f$                                                                                                                                                                    | <p>The total lift of the wing is</p> $L = \frac{1}{2} \rho V^2 S C_L$ $= \frac{1}{2} (0.002378) (168.8)^2 (10) (0.7480)$ $= 253.4 \text{ lb}_f$                                                                                                                                                                                   |
| <p>Induced drag is found from the standard relation</p> $C_{Di} = \frac{C_L^2}{\pi \cdot AR} = \frac{(0.8846)^2}{\pi \cdot (10)} = 0.02491$ $D_i = \frac{1}{2} \rho V^2 S C_{Di}$ $= \frac{1}{2} (0.002378) (168.8)^2 (10) (0.02491)$ $= 8.4 \text{ lb}_f$                                                         | <p>Induced drag is found from the standard relation</p> $C_{Di} = \frac{C_L^2}{\pi \cdot AR} = \frac{(0.7480)^2}{\pi \cdot (10)} = 0.01781$ $D_i = \frac{1}{2} \rho V^2 S C_{Di}$ $= \frac{1}{2} (0.002378) (168.8)^2 (10) (0.01781)$ $= 6.0 \text{ lb}_f$                                                                        |
| <p>Lift to drag ratio: <math>\frac{L}{D} = \frac{0.8846}{0.02491} = 35.5</math></p>                                                                                                                                                                                                                                | <p>Lift to drag ratio: <math>\frac{L}{D} = \frac{0.7480}{0.01781} = 42.0</math></p>                                                                                                                                                                                                                                               |

### V2.3 Results from SURFACES

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Summary for wing with 0° leading edge sweep\*:

| Parameter                | Symbol        | Classic Method      | SURFACES            |
|--------------------------|---------------|---------------------|---------------------|
| Lift curve slope         | $C_{L\alpha}$ | 0.0885              | 0.0860              |
| Lift coefficient         | $C_L$         | 0.885               | 0.845               |
| Induced drag coefficient | $C_{Di}$      | 0.0249              | 0.0227              |
| Lift force               | $L$           | 300 lb <sub>f</sub> | 286 lb <sub>f</sub> |
| Induced drag force       | $D_i$         | 8.4 lb <sub>f</sub> | 7.7 lb <sub>f</sub> |
| Lift-to-drag ratio       | $L/D_i$       | 35.5                | 37.2                |

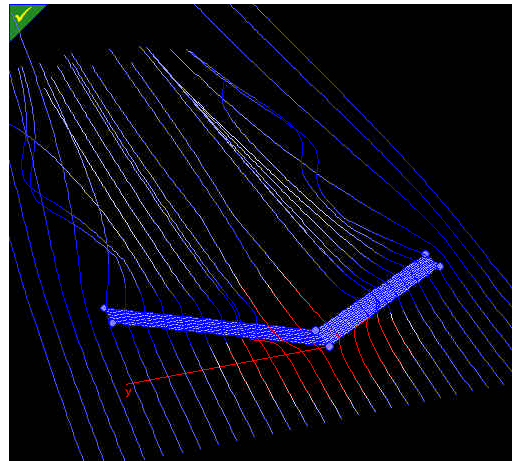
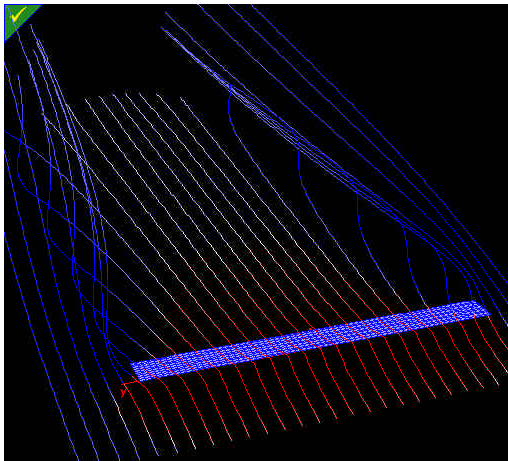
\*Span efficiency for both cases is unrealistically assumed to be 1.

Summary for wing with 35° leading edge sweep\*:

| Parameter                | Symbol        | Classic Method      | SURFACES            |
|--------------------------|---------------|---------------------|---------------------|
| Lift curve slope         | $C_{L\alpha}$ | 0.0748              | 0.07365             |
| Lift coefficient         | $C_L$         | 0.748               | 0.723               |
| Induced drag coefficient | $C_{Di}$      | 0.0178              | 0.0166              |
| Lift force               | $L$           | 253 lb <sub>f</sub> | 245 lb <sub>f</sub> |
| Induced drag force       | $D_i$         | 6.0 lb <sub>f</sub> | 5.6 lb <sub>f</sub> |
| Lift-to-drag ratio       | $L/D_i$       | 42.0                | 43.5                |

\*Span efficiency for both cases is unrealistically assumed to be 1.

Printout from **SURFACES**:



## Validation 3 - Warren 12 Wing

### V3.1 Model

The Warren-12 wing is a standard Vortex-Lattice model used to check the accuracy of vortex lattice codes. It provides a ready check case for the evaluation of any new or modified code, as well as a check on the panel scheme layout. This wing is known as the Warren 12 planform, and is defined, together with the “official” characteristics from previous calculations, in Fig. 3-1 below.

For the results cited, the reference chord used in the moment calculation is the average chord (slightly nonstandard, normally the reference chord used is the mean aerodynamic chord) and the moment reference point is located at the wing apex (which is also nonstandard).

“Published” Data:

AR = 2.83  
 $\Lambda_{LE} = 53.54^\circ$   
 $C_{ref} = 1.00$   
 $X_{CG} = 0.00$   
 $S_{wing} = 2.83$   
 $C_{L\alpha} = 2.743 / \text{rad}$   
 $C_{M\alpha} = -3.10 / \text{rad}$

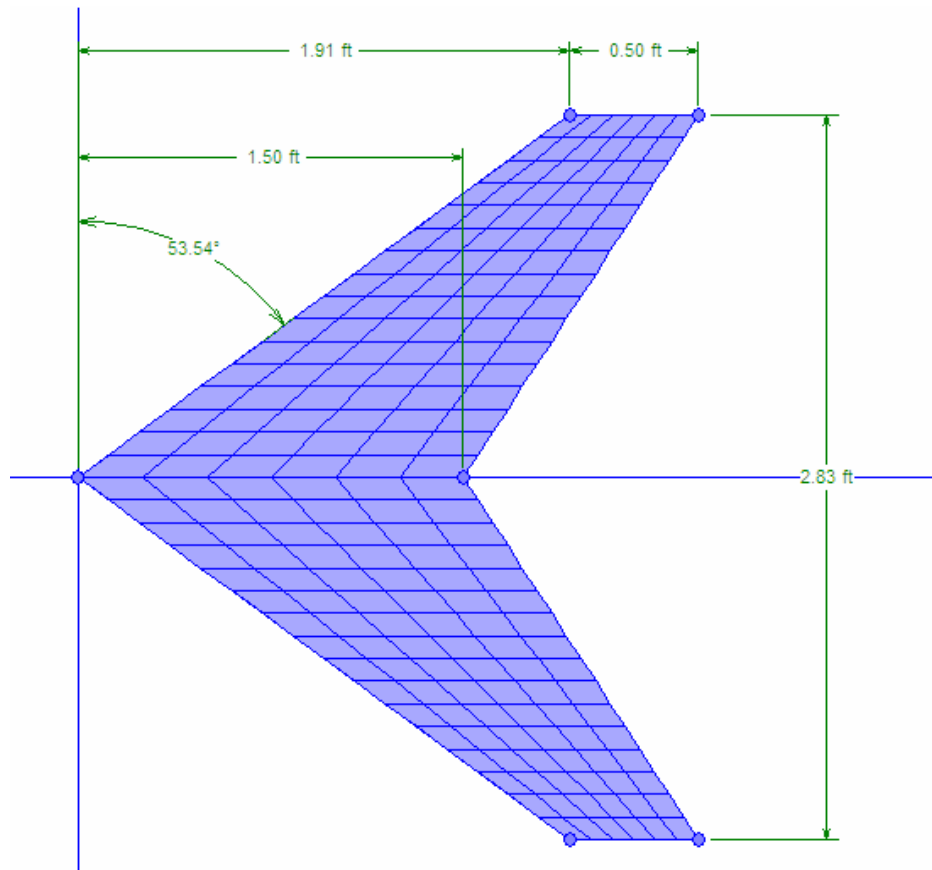


Figure 3-1: Warren-12 planform

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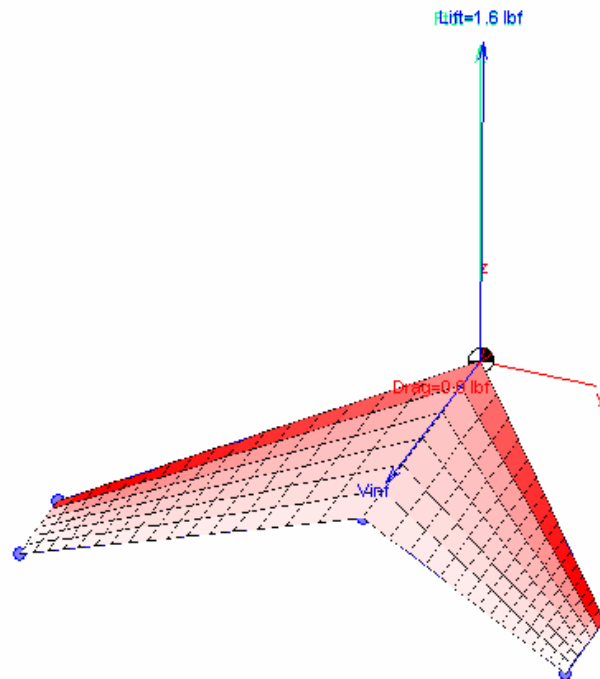


**Figure 3-2: Warren-12 planform VL results**

### **V3.2 Expected Result**

The following results are expected:

$$C_{L\alpha} = 2.743 / \text{rad}$$

$$C_{M\alpha} = -3.10 / \text{rad}$$

### **V3.3 Results from SURFACES**

The following results were obtained from **SURFACES** for 6 chordwise by 16 spanwise panels on each wing (total of 192 panels):

$$C_{L\alpha} = 2.790 / \text{rad}$$

$$C_{M\alpha} = -3.174 / \text{rad}$$

The following results were obtained from **SURFACES** for 8 chordwise by 24 spanwise panels on each wing (total of 384 panels):

$$C_{L\alpha} = 2.776 / \text{rad}$$

$$C_{M\alpha} = -3.152 / \text{rad}$$

The following results were obtained from **SURFACES** for 16 chordwise by 36 spanwise panels on each wing (total of 1296 panels):

$$C_{L\alpha} = 2.767 / \text{rad}$$

$$C_{M\alpha} = -3.139 / \text{rad}$$

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## Validation 4 - Bertin-Smith 2-D Wing

### V4.1 Model

Calculations for a highly swept back, high aspect ratio wing is provided in the text *Aerodynamics for Engineers* by Bertin and Smith. This wing has detailed calculations shown in Example 6-2 (page 198) in the text. The model in the text was recreated using **SURFACES**. Additionally, a comparison to another VLM code (*Tornado*) is made.

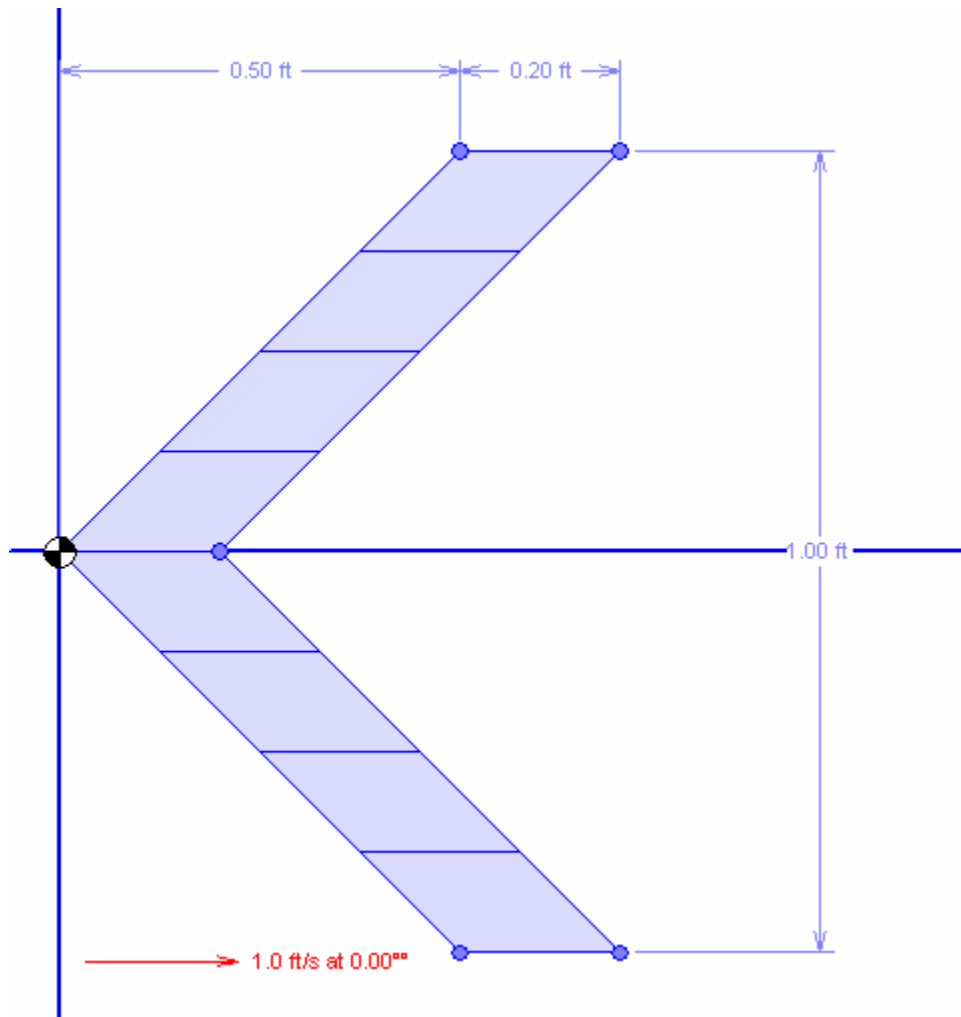


Figure 4-1: The Bertin-Smith swept back wing.

### V4.2 Expected Result

Is obtained from the book *Aerodynamics for Engineers*, by Bertin and Smith. The data is obtained from the calculations on page 202, but the resulting lift curve slope is:

$$C_{L\alpha} = 0.05992 / ^\circ = 3.433 / \text{rad}$$

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### V4.3 Results from SURFACES

The following results were obtained from **SURFACES** for 1 chordwise by 4 spanwise panels on each wing (total of 8 panels):

$$C_{L\alpha} = 0.06011 / ^\circ = 3.442 / \text{rad}$$

**SURFACES** yields a difference of 0.26%. Another VLM code, called *Tornado*, considers the same problem. In his Master Thesis, "A Vortex Lattice MATLAB Implementation for Linear Aerodynamic Wing Applications" the author of *Tornado*, Mr. Tomas Melin, reports a lift curve slope of 3.450 /rad using *Tornado*. The difference using that code is 0.5%.

It can be seen that both codes are very close to the theoretical calculations in the source, but **SURFACES** yields less difference than *Tornado*. It should also be noted that the calculations in the source only carries 4 significant digits through the calculations – **SURFACES** uses a double floating point accuracy.

Summary:

| Parameter        | Symbol        | Bertin-Smith | TORNADO       | SURFACES      |
|------------------|---------------|--------------|---------------|---------------|
| Lift curve slope | $C_{L\alpha}$ | 3.433        | 3.450 (0.50%) | 3.442 (0.26%) |

## Validation 5 - Cessna 172

### Comparison of Several Codes

#### V5.1 Model

A model of the Cessna 172 was constructed to compare stability derivatives from **SURFACES** to other VLM codes (AVL, VIRGIT, TORNADO) and the panel code CMARC, as well as published Cessna data. The model has the camber line of the NACA 2412 airfoil of the Cessna 172. Additionally, it has a  $1^{\circ}30'$  angle-of-incidence at the root of the wing and  $-1^{\circ}30'$  at the tip, and a  $1^{\circ}44'$  dihedral like the original airplane. A sweep of parameters was performed at an airspeed of 178.9 ft/s, at an altitude of 4921 ft ( $\rho = 0.002054$  slugs/ft<sup>3</sup>), and at a weight of 2207 lbs.

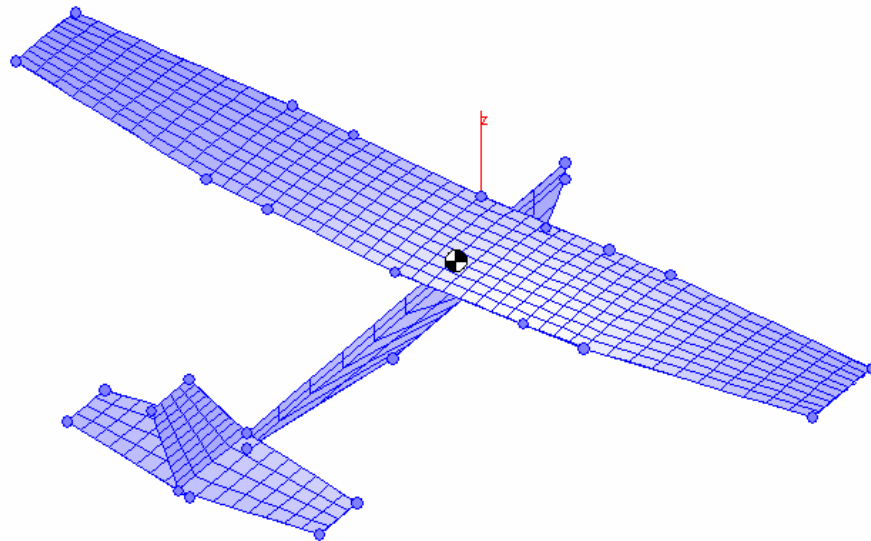


Figure 5-1: A Model of the C-172

#### V5.2 Expected Result

##### Range for $C_{L_a}$ :

The following parameters are given:

Wing area  $S = 174$  ft<sup>2</sup>  
Aspect Ratio  $AR = b^2 / S = 36.08^2 / 174 = 7.48$

Assume a 2-D lift curve slope of  $C_{L_{\alpha 2D}} = 0.107$  per deg (for NACA 2412, from *Theory of Wing Sections*, by Abbott and Doenhoff, page 478).

Compute a 3-D lift curve slope from Method 1 of USAF DATCOM Section 1, page 1-7.

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$$C_{L_{\alpha}} = \frac{2\pi \cdot AR}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{C/2}}{\beta^2} \right)} + 4}$$

Where; AR = Wing Aspect Ratio = 7.48

$\beta$  = Mach number parameter (Prandtl-Glauert) =  $(1-M^2)^{0.5} \approx 1$

$\kappa$  = Ratio of 2-D lift curve slope to  $2\pi = 0.107 \times (180/\pi)/(2\pi) = 0.97572$

$\Lambda_{C/2}$  = Sweepback of mid-chord =  $0^\circ$

$$\begin{aligned} C_{L_{\alpha}} &= \frac{2\pi \cdot AR}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{C/2}}{\beta^2} \right)} + 4} \\ &= \frac{2\pi \cdot 7.48}{2 + \sqrt{\frac{7.48^2}{0.97572^2} (1) + 4}} = 4.74 \text{ per rad} = 0.08267 \text{ per deg} \end{aligned}$$

#### Range for $C_{n_{\beta}}$ :

Consider the following check for  $C_{n_{\beta}}$ . The height, root, and tip chord of the fin is 5.50 ft, 4.25 ft, and 2.30 ft, respectively. The leading edge sweep is  $40^\circ$ . The airfoil is a NACA 0009 airfoil, whose properties are discussed in Validation Sample 2. Using this data we compute the following lift curve slope for the fin:

Fin area  $S_{fin} = \frac{1}{2} \cdot (4.25 + 2.30) \cdot 5.50 = 18.01 \text{ ft}^2$

Aspect Ratio  $AR = b_{fin}^2 / S_{fin} = 5.50^2 / 18.01 = 1.679$

Assume a 2-D lift curve slope of  $C_{L_{\alpha 2D}} = 0.1063$  per deg (for NACA0009, from *Theory of Wing Sections*, by Abbott and Doenhoff).

Compute a 3-D lift curve slope from Method 1 of USAF DATCOM Section 1, page 1-7.

$$C_{L_{\alpha}} = \frac{2\pi \cdot AR}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{C/2}}{\beta^2} \right)} + 4}$$

Where;  $\beta$  = Mach number parameter (Prandtl-Glauert) =  $(1-M^2)^{0.5} \approx 1$

$\kappa$  = Ratio of 2D lift curve slope to  $2\pi = 0.1063 \times (180/\pi)/(2\pi) = 0.96934$

$\Lambda_{C/2}$  = Sweepback of mid-chord  $\approx 28^\circ$

$$\begin{aligned}
 C_{L\alpha} &= \frac{2\pi \cdot AR}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{C/2}}{\beta^2} \right) + 4}} \\
 &= \frac{2\pi \cdot 1.679}{2 + \sqrt{\frac{1.679^2}{0.96934^2} \left( 1 + \frac{\tan^2(28^\circ)}{1^2} \right) + 4}} = \\
 &= \frac{10.54947}{2 + \sqrt{3.00019 \times 1.28271 + 4}} = 2.197 \text{ per rad} = 0.03835 \text{ per deg}
 \end{aligned}$$

If one considers the fin at a  $\beta = 1^\circ$ , the fin lift coefficient is given by  $C_L = 1^\circ \times C_{L\alpha} = 0.03835$ . The total lift of the fin at  $V = 178.9$  ft/s and  $\rho = 0.002054$  slugs/ft<sup>3</sup> is found to be

$$L_{fin} = \frac{1}{2} \rho V^2 S C_L = \frac{1}{2} (0.002054) (178.9)^2 (18.01) (0.03835) = 22.7 \text{ lb}_f$$

Assuming a tail arm from reference point of 16.0 ft, the total moment is found to be 363.2 ft·lb<sub>f</sub>, which yields a  $C_n$  of:

$$C_n = \frac{N}{\frac{1}{2} \rho V^2 S b} = \frac{363.2}{\frac{1}{2} \times 0.002054 \times 178.9^2 \times 174 \times 36.17} = 0.00176$$

Since  $N$  equals 0 ft·lb<sub>f</sub> at  $\beta = 0^\circ$ ,  $C_{n\beta}$  can be found to be:

$$C_{n\beta} = \frac{\partial C_n}{\partial \beta} \approx \frac{0.00176}{1^\circ} = 0.00176 \text{ per } ^\circ = 0.1006 \text{ per rad}$$

From this, a reasonable  $C_{n\beta}$  for this plane should be of the order of 0.03-0.17, depending on the contribution of other components of the airplane.

### V5.3 Results from SURFACES

The following results were obtained from **SURFACES** and compared to that of other VLM codes. The data is obtained from the *Tornado* manual, pages 34-38. All the stability derivatives presented below are evaluated at  $\alpha = 0$ .

**TABLE 5-1: Stability Derivatives at  $\alpha = 0$ :**

|               | TEST <sup>1</sup> | AVL  | VIRGIT | CMARC | TORNADO | SURFACES     | NOTE |
|---------------|-------------------|------|--------|-------|---------|--------------|------|
| $C_{L\alpha}$ | 4.6               | 4.98 | 5.25   | 5.214 | 5.2763  | 5.128/5.1803 | 1    |
| $C_{D\alpha}$ | 0.13              | 0    | -0.005 | 0.086 | -0.022  | 0.051/0.146  | 2    |
| $C_{Y\alpha}$ | -                 | 0    | 0      | 0     | 0       | 0            | -    |
| $C_{l\alpha}$ | -                 | 0    | 0      | 0     | 0       | 0            | -    |

<sup>1</sup> Comparison data is obtained from *Airplane Flight Dynamics and Automatic Flight Controls*, by Jan Roskam. Appendix C, page 592.

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|               |       |       |       |        |        |         |   |
|---------------|-------|-------|-------|--------|--------|---------|---|
| $C_{m\alpha}$ | -0.89 | -0.33 | -0.85 | -1.432 | -1.498 | -1.148/ | 3 |
| $C_{n\alpha}$ | -     | 0     | 0     | 0      | 0      | 0       | - |

|              |        |       |       |        |       |                |   |
|--------------|--------|-------|-------|--------|-------|----------------|---|
| $C_{L\beta}$ | -      | 0     | 0     | 0      | 0     | 0              | - |
| $C_{D\beta}$ | -      | 0     | 0     | 0      | 0     | 0              | - |
| $C_{Y\beta}$ | -0.31  | -0.26 | -0.24 | -0.104 | -0.3  | -0.370/-0.341  | 4 |
| $C_{l\beta}$ | -0.089 | 0.33  | 0.007 | 0.063  | 0.025 | -0.0479/-0.045 | 5 |
| $C_{m\beta}$ | -      | 0     | 0     | 0      | 0     | 0              | - |
| $C_{n\beta}$ | 0.065  | 0.092 | 0.1   | 0.042  | 0.12  | 0.117/0.0911   | 6 |

|          |       |        |       |        |        |               |   |
|----------|-------|--------|-------|--------|--------|---------------|---|
| $C_{Lp}$ | -     | 0      | 0     | 0      | 0      | 0             | - |
| $C_{Dp}$ | -     | 0      | 0     | 0      | 0      | 0             | - |
| $C_{Yp}$ | -     | -0.066 | -0.1  | -0.015 | -0.039 | -0.110        | 7 |
| $C_{lp}$ | -0.47 | -0.325 | -0.52 | -0.995 | -0.526 | -0.510/-0.508 | 7 |
| $C_{mp}$ | -     | 0      | 0     | 0      | 0      | 0             | - |
| $C_{np}$ | -0.03 | -0.007 | -0.01 | -0.133 | -0.006 | -0.0056/0.018 | 7 |

|          |       |        |     |         |        |              |       |
|----------|-------|--------|-----|---------|--------|--------------|-------|
| $C_{Lq}$ | -     | 9.41   | 9.3 | 9.003   | 10.18  | 7.894/9.111  | 8, 9  |
| $C_{Dq}$ | -     | 0      | 0   | 0       | 0.128  | 0.432/0.256  | 8, 10 |
| $C_{Yq}$ | -     | 0      | 0   | 0       | 0      | 0            | -     |
| $C_{lq}$ | -     | 0      | 0   | 0       | 0      | 0            | -     |
| $C_{mq}$ | -12.4 | -14.43 | -15 | -17.155 | -14.96 | -12.156/14.3 | 8, 9  |
| $C_{nq}$ | -     | 0      | 0   | 0       | 0      | 0            | -     |

|          |        |        |        |        |       |               |      |
|----------|--------|--------|--------|--------|-------|---------------|------|
| $C_{Lr}$ | -      | 0      | 0      | 0      | 0     | 0/0.0029      | -    |
| $C_{Dr}$ | -      | 0      | 0      | 0      | 0     | 0             | -    |
| $C_{Yr}$ | 0.21   | 0.209  | 0.23   | 0.45   | 0.271 | 0.296/0.306   | 7, 9 |
| $C_{lr}$ | 0.096  | 0.021  | 0.008  | 0.195  | 0.009 | 0.101/0.0926  | 7, 9 |
| $C_{mr}$ | -      | 0      | 0      | 0      | 0     | 0             | -    |
| $C_{nr}$ | -0.099 | -0.075 | -0.095 | -0.212 | -0.11 | -0.115/-0.119 |      |

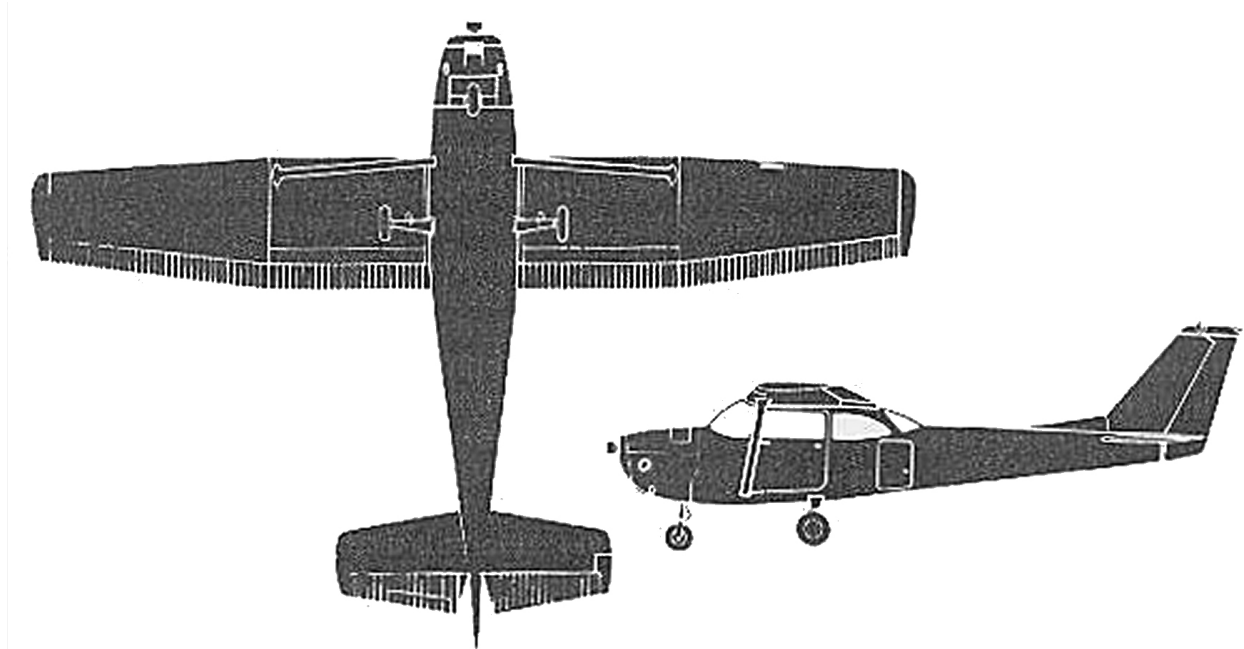
All derivatives are per radian.

#### NOTES:

- (1) There is a known difference in input geometry, which will likely cause numerical discrepancies. It is not known if the other VLM codes included washout, dihedral, and wing camber like the **SURFACES** model.
- (2) A value of zero is expected at  $C_L = 0$  only if the airfoil of the wing is symmetrical (flat plate).
- (3) The different values are primarily due to the different reference locations, but also due to possible power effects. For instance, **SURFACES** and *VIRGIT* use 29.5% of MAC, *Tornado* uses 31.9% MAC. *AVL* and *CMARC* reference points are unknown. **SURFACES** has the reference point located 2 ft below the wing plane and does not account for power effects – it is unknown where the other codes place the vertical location of the reference point, or if propeller normal force is accounted for.
- (4) Note that for **SURFACES** the standard coordinate system is used with the Angle-of-Yaw (positive beta) coming from the left, rather than the right. Consequently, a sign change is added to compare to the other codes.

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- (5) **SURFACES** evaluated a restoring dihedral effect for the C-172 – the only one of the above codes.
- (6) [Deleted]
- (7) The rate of roll and yaw derivatives are obtained with respect to  $P \cdot B_{ref} / (2 \cdot V_{inf})$ . For that reason, derivatives with respect to P or R are multiplied by the factor  $2 \cdot V_{inf} / B_{ref}$ .
- (8) The rate of pitch derivative is obtained with respect to  $P \cdot C_{ref} / (2 \cdot V_{inf})$ . For that reason, derivatives with respect to Q are multiplied by the factor  $2 \cdot V_{inf} / C_{ref}$ .
- (9) Differences are most likely due to modeling differences and differences in location of reference point.
- (10) A change in lift should be associated with a change in drag. It is not known why *Tornado* and **SURFACES** are the only codes to display a value here.



#### V5.4 Comparison of Codes

Table 5-1 prompts some interesting questions – for instance, how do the codes compare? Table 5-2 displays one such comparison. Here, a grade from 1 (worst) to 5 (best) is assigned to those stability parameters that can be compared to the source. The parameters are compared by computing difference using:

$$\Delta difference = \left| \frac{P_{CODE} - P_{SOURCE}}{P_{SOURCE}} \right|$$

Then, the code with the largest difference scores 1 and the code with the smallest one 5. A total of 30 derivatives are considered in Table 5-1, of which 12 have a value from the source document (*Airplane Flight Dynamics and Automatic Flight Controls*, by Jan Roskam). The highest total score a code can receive is  $5 \times 12 = 60$ . The lowest total score is 12. The scores for the 5 codes are compared in Table 5-2:

**Table 5-2: Comparison of Several VLM Codes and the Panel Code CMARC.**

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|                                  | AVL       | VIRGIT    | CMARC     | TORNADO   | SURFACES  |
|----------------------------------|-----------|-----------|-----------|-----------|-----------|
| $C_{L\alpha}$                    | 5         | 2         | 3         | 1         | 4         |
| $C_{D\alpha}$                    | 3         | 2         | 5         | 1         | 4         |
| $C_{m\alpha}$ ( $CM_{y\alpha}$ ) | 2         | 5         | 3         | 1         | 4         |
| $C_{y\beta}$                     | 4         | 2         | 1         | 5         | 3         |
| $C_{l\beta}$ ( $CM_{x\beta}$ )   | 1         | 4         | 2         | 3         | 5         |
| $C_{n\beta}$ ( $CM_{z\beta}$ )   | 4         | 3         | 5         | 1         | 2         |
| $C_l, P$ ( $CM_x, P$ )           | 2         | 4         | 1         | 3         | 5         |
| $C_n, P$ ( $CM_z, P$ )           | 4         | 5         | 1         | 3         | 2         |
| $C_m, Q$ ( $CM_y, Q$ )           | 4         | 2         | 1         | 3         | 5         |
| $C_y, R$                         | 5         | 4         | 1         | 3         | 2         |
| $C_l, R$ ( $CM_x, R$ )           | 4         | 2         | 1         | 3         | 5         |
| $C_n, R$ ( $CM_z, R$ )           | 2         | 5         | 1         | 4         | 3         |
| <b>TOTAL SCORE</b>               | <b>40</b> | <b>40</b> | <b>25</b> | <b>31</b> | <b>44</b> |
| Number of 1s                     | 1         | 0         | 7         | 4         | 0         |
| Number of 2s                     | 3         | 5         | 1         | 0         | 2         |
| Number of 3s                     | 1         | 1         | 1         | 6         | 2         |
| Number of 4s                     | 5         | 3         | 0         | 1         | 3         |
| Number of 5s                     | 2         | 3         | 2         | 1         | 4         |

Table 5-2 shows that **SURFACES** scores highest (44 points). CMARC scored worst (25 points). Two codes never scored worse than 2, VIRGIT and **SURFACES**. On the other hand, AVL, CMARC, and TORNADO all have at least one worst score. The most frequent low score (1) was received by CMARC, 7 times. The most frequent high score (5) was received by **SURFACES**, 4 times. In fact, **SURFACES** was the only code to correctly compute a restoring dihedral effect for the Cessna 172.

## Validation 6 - 2-D $C_L$ , $C_D$ , $C_M$ for NACA 23012

### V6.1 Model

A high aspect ratio ( $AR=20$ ) wing model was constructed to perform a 3-D similarity evaluation to a standard 3-D aerodynamic analysis. The model has a wing span of 20 ft and a chord of 1 ft. An angle sweep of attack from  $-8^\circ$  through  $8^\circ$  at an airspeed of 100 ft/sec and density of  $0.002378$  slugs/ft<sup>3</sup> was performed. The model has 16 chordwise and 60 spanwise panels. The panels form the camber line of the NACA 23012 airfoil. The purpose of this validation is to demonstrate how **SURFACES** simulates airfoil properties.

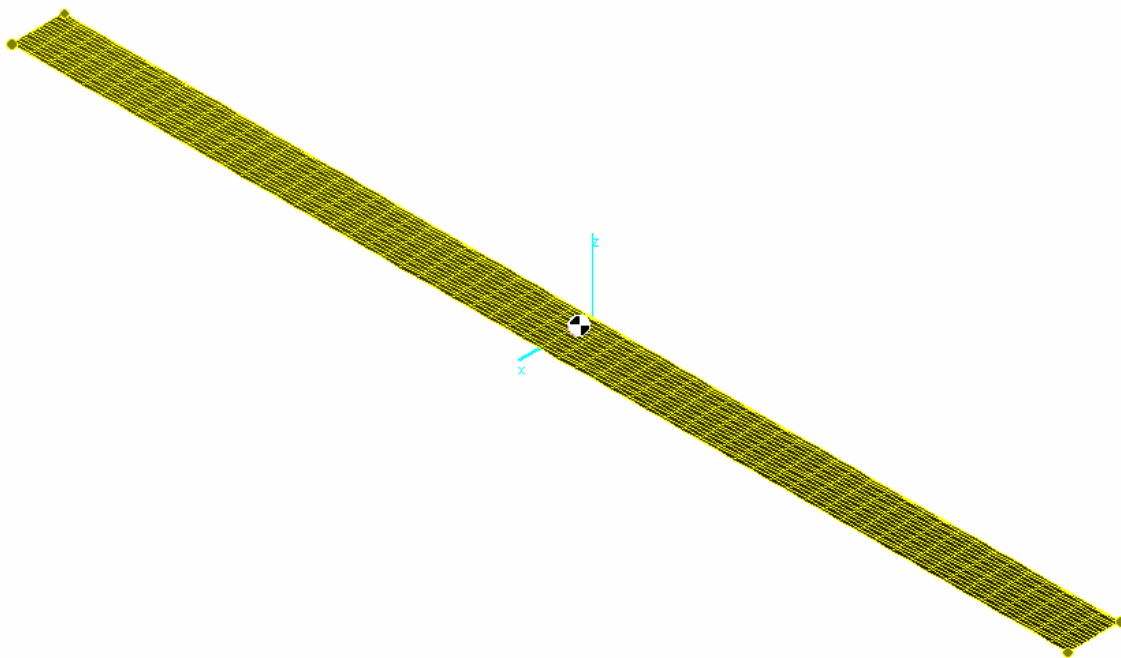


Figure 6-1: 3-D wing model with a 23012 airfoil

### V6.2 Expected Result

The following parameters are given:

Airspeed  $V = 100$  ft/s  
Wing area  $S = 20 \times 1 = 20$  ft<sup>2</sup>  
Aspect Ratio  $AR = b^2 / S = 20^2 / 10 = 20$

The 2-D lift curve slope of  $C_{l_\alpha} = 0.1051$  per deg,  $C_{l_0} = 0.1233$ , and  $C_{m_\alpha} = 0.00020\alpha - 0.01198$  is obtained from interpolation (for NACA 23012, from *Theory of Wing Sections*, by Abbott and Doenhoff).

Compute a 3-D lift curve slope from Method 1 of USAF DATCOM Section 1, page 1-7.

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$$C_{L_\alpha} = \frac{AR \cdot C_{l_\alpha}}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{C/2}}{\beta^2} \right) + 4}}$$

Where; AR = Wing Aspect Ratio = 20

$\beta$  = Mach number parameter (Prandtl-Glauert) =  $(1-M^2)^{0.5} \approx 1$

$\kappa$  = Ratio of 2D lift curve slope to  $2\pi = 0.1051 \times (180/\pi)/(2\pi) = 0.95840$

$\Lambda_{C/2}$  = Sweepback of mid-chord =  $0^\circ$

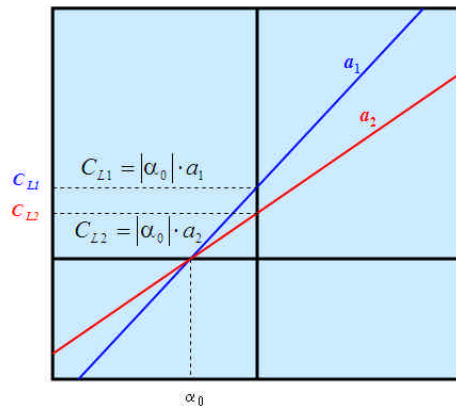
$$\begin{aligned} C_{L_\alpha} &= \frac{2\pi \cdot AR}{2 + \sqrt{\frac{AR^2 \beta^2}{\kappa^2} \left( 1 + \frac{\tan^2 \Lambda_{C/2}}{\beta^2} \right) + 4}} \\ &= \frac{2\pi \cdot 20}{2 + \sqrt{\frac{400}{0.95840^2} \left( 1 + \frac{0}{1^2} \right) + 4}} = 5.472 \text{ per rad} = 0.09551 \text{ per deg} \end{aligned}$$

Compute zero lift angle for the 2D airfoil using:  $C_{l_0} = -\alpha_0 \cdot C_{l_\alpha} \Leftrightarrow \alpha_0 = -\frac{C_{l_0}}{C_{l_\alpha}} = -\frac{0.1233}{0.1051} = -1.173$

Compute lift at zero angle for the 3D wing using:  $C_{L_0} = -\alpha_0 \cdot C_{L_\alpha} = -(-1.173) \cdot 0.09551 = 0.1121$

Compute pitching moment for 3D wing:

$$\begin{aligned} C_{m_\alpha} &= C_{l_\alpha} \cdot \Delta x \Leftrightarrow \Delta x = \frac{C_{m_\alpha}}{C_{l_\alpha}} \\ (C_{m_\alpha})_{3D} &= C_{L_\alpha} \cdot \Delta x = C_{L_\alpha} \cdot \left( \frac{C_{m_\alpha}}{C_{l_\alpha}} \right) \\ &= (0.09551) \cdot \left( \frac{-0.01198}{0.1051} \right) \\ &= -0.01089 \end{aligned}$$



### V6.3 Results from SURFACES

Summary:

| Parameter                  | Symbol         | Experiment <sup>2</sup>          | Classical Method | Surfaces                         |
|----------------------------|----------------|----------------------------------|------------------|----------------------------------|
| Lift curve slope           | $C_{L_\alpha}$ | 0.1051 (2-D value)               | 0.0955           | 0.0943                           |
| Lift coefficient intercept | $C_{L_0}$      | 0.1233                           | 0.1121           | 0.1194                           |
| Moment coefficient         | $C_{M_\alpha}$ | $0.00020 \cdot \alpha - 0.01198$ |                  | $0.00016 \cdot \alpha - 0.01888$ |

<sup>2</sup> Theory of Wing Sections, by Abbott & Doenhoff, graph on page 498.

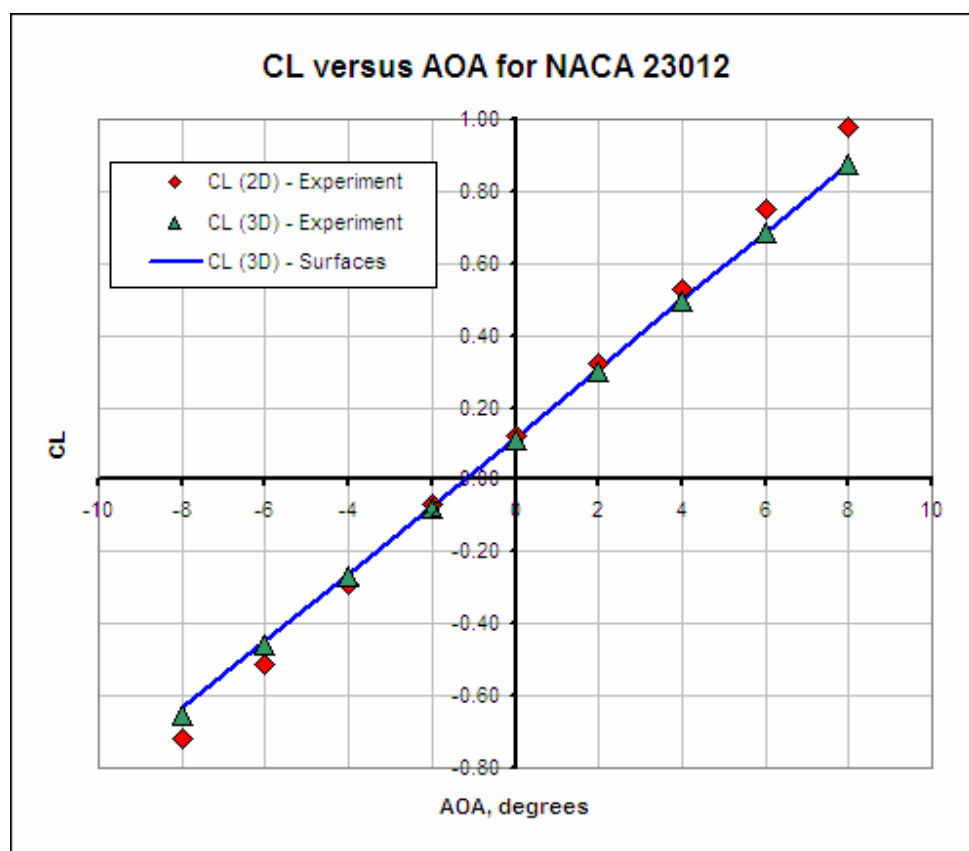


Figure 6-2: 3-D wing model with a 23012 airfoil camber line.

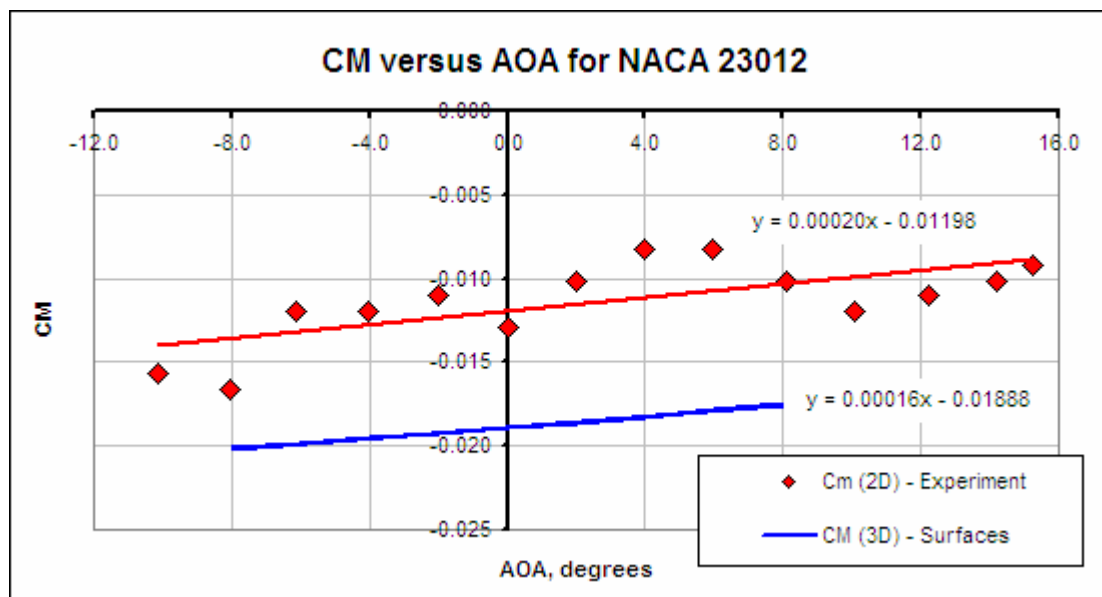


Figure 6-3: 3-D wing model with a 23012 airfoil camber line.

## Validation 7 - F-104 Starfighter

### V7.1 Model

A model of the Lockheed F-104 Starfighter was constructed to compare selected stability derivatives from **SURFACES** to that presented in the text *Flight Stability and Automatic Control*, by Robert C. Nelson. The data can be found in Appendix B of the text, on page 253.

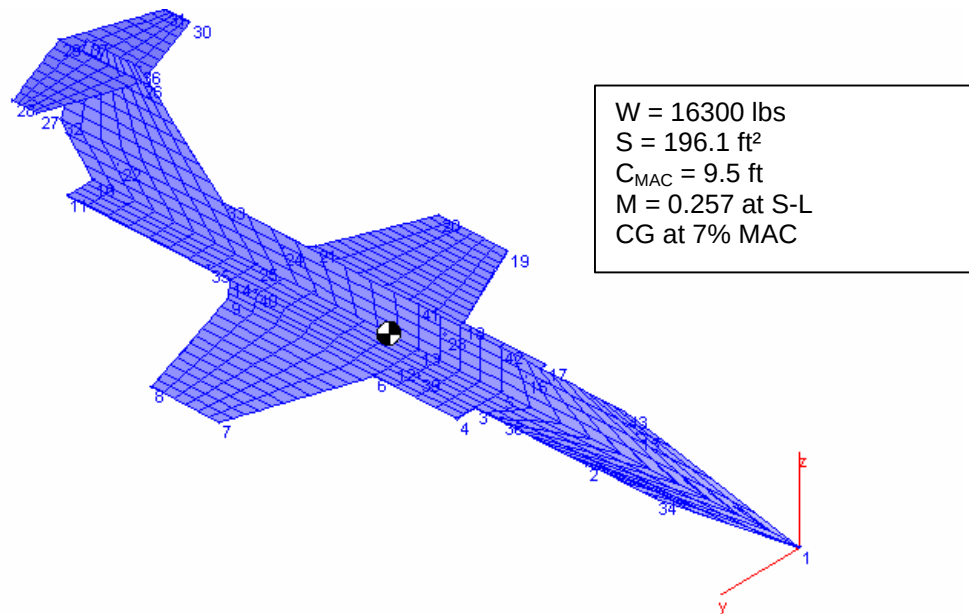


Figure 7-1: 3-D Vortex-Lattice model of the F-104 Starfighter

### V7.2 Results from SURFACES

Summary:

| Parameter                            | Symbol        | Source <sup>3</sup> | SURFACES           | %Difference |
|--------------------------------------|---------------|---------------------|--------------------|-------------|
| Lift coefficient                     | $C_L$         | 0.735               | 0.717 <sup>4</sup> | 2.4%        |
| Drag coefficient                     | $C_D$         | 0.263               | 0.175 <sup>5</sup> | 33.5%       |
| Lift curve slope                     | $C_{L\alpha}$ | 3.44                | 3.36               | 2.4%        |
| Drag curve slope                     | $C_{D\alpha}$ | 0.45                | 0.66 <sup>6</sup>  | 45.8%       |
| Moment slope (CM $\alpha$ )          | $C_{M\alpha}$ | -0.64               | -0.756             | 18.1%       |
| Side force slope (CF $\beta$ )       | $C_{Y\beta}$  | -1.17               | -1.12              | 4.3%        |
| Dihedral effect (CM $\beta$ )        | $C_{l\beta}$  | -0.175              | -0.156             | 10.9%       |
| Weathercock stability (CM $z\beta$ ) | $C_{n\beta}$  | 0.50                | 0.491              | 1.8%        |

All derivatives are per radian. At M=0.257

<sup>3</sup> *Flight Stability and Automatic Control*, by Robert C. Nelson.

<sup>4</sup> Note that  $V = 0.257 \times 1116 \text{ ft/s} = 286.8 \text{ ft/s}$ . Therefore, Lift is  $\frac{1}{2} \cdot 0.002378 \cdot 286.8^2 \cdot 196.1 \cdot 0.735 = 14097 \text{ lbf}$ . This is the same lift SURFACES generated to get the given lift coefficient.

<sup>5</sup> Using the *surface integration* method

<sup>6</sup> This is highly dependent on drag model. Here,  $C_{D\alpha} = (0.0009474 \cdot \alpha - 0.0004737) \cdot 180 / \pi$ , which at  $\alpha = 12.6$  becomes 0.656.

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**Figure 7-2: A Starfighter in flight**

Image from <http://www.starfighters.net/gallery/1999gallery/1999gallery.html>

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## Validation 8 - Ryan Navion

### V8.1 Model

A model of the Ryan Navion was constructed compare to the analysis of Example Problem 2.1 found in Robert C. Nelsons "*Flight Stability and Automatic Control*", on pages 53-58. The VL model was based on the three-view in Figure 8-1.

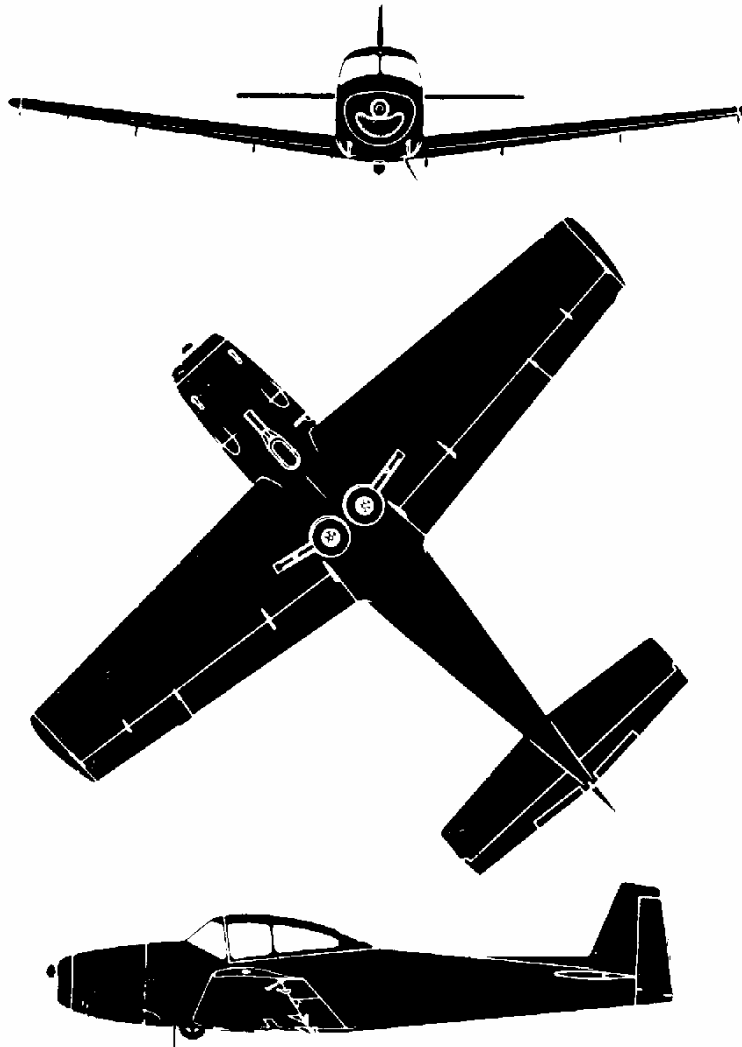


**Figure 8-1. A three-view drawing of the Ryan Navion.**

The reference document determines several parameters for the Navion in Problem 2.1. The calculation of selected parameters is repeated in Section V8.1 for convenience.

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Figure 8-2. A Ryan Navion in flight – Photographer unknown.

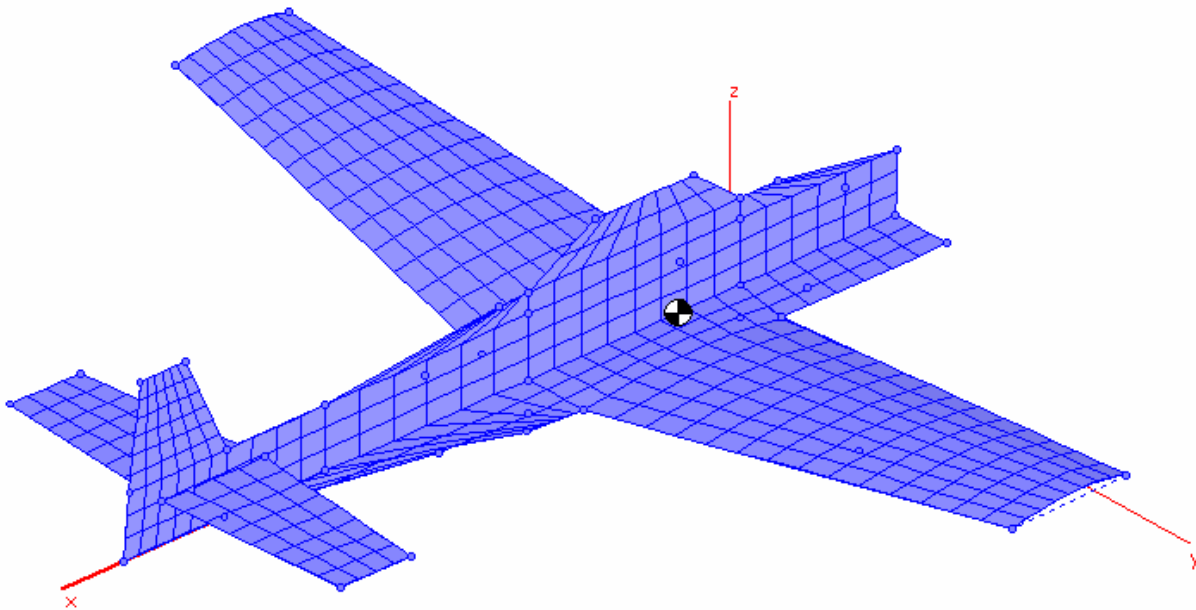


Figure 8-3. The **SURFACES** Vortex-Lattice model of the Ryan Navion.

### V8.2 Expected Result

The stick-fixed neutral point is estimated from Equation (2.37) in the reference document, here written using variables more consistent with this document:

$$\frac{X_{NEU}}{C_{REF}} = \frac{X_{AC}}{C_{REF}} - \frac{C_{M_{\alpha f}}}{C_{L_{\alpha w}}} + \eta V_{HT} \frac{C_{L_{\alpha t}}}{C_{L_{\alpha w}}} \left( 1 - \frac{d\varepsilon}{d\alpha} \right)$$

Where (note that numerical values are obtained from the document);

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$C_{REF}$  = Reference wing chord = 5.7 ft  
 $C_{L\alpha w}$  = Slope of wing lift coefficient = 4.3 per rad  
 $C_{L\alpha t}$  = Slope of HT lift coefficient = 3.91 per rad  
 $C_{M\alpha f}$  = Slope of fuselage moment coefficient = 0.12 per rad  
 $d\epsilon/d\alpha$  = Variation of downwash with angle-of-attack = 0.45  
 $V_{HT}$  = Horizontal tail volume = 0.66  
 $X_{AC}$  = Aerodynamic center of wing-body combination = 1.425 ft  
 $\eta$  = Tail efficiency = 1

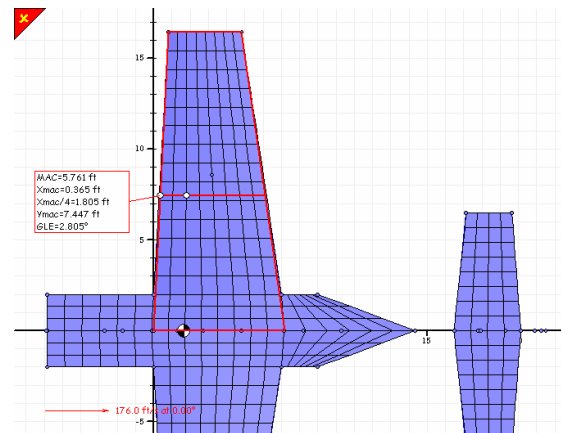
$$\frac{X_{NEU}}{C_{REF}} = \frac{1.425}{5.7} - \frac{0.12}{4.3} + (1)(0.66)\frac{3.91}{4.3}(1 - 0.45) = 0.552$$

Note that the reference document (which is a First Edition) states the  $X_{neu}$  is at 0.37, but in conversation with the author (R. C. Nelson) it was confirmed this was an error that had been corrected for later editions of the book.

Note that the planform properties of the VL model were determined using **SURFACES'** built in tool, which printed out the following analysis report:

#### MEAN AERODYNAMIC CHORD ANALYSIS

|                                    |                               |
|------------------------------------|-------------------------------|
| Surface chord, root .....          | Cr = 7.200 ft                 |
| Surface LE, root .....             | Xr = 0.000 ft                 |
| .....                              | Yr = 0.000 ft                 |
| Surface chord, tip .....           | Ct = 4.022 ft                 |
| Surface LE, tip .....              | Xr = 0.806 ft                 |
| .....                              | Yr = 16.446 ft                |
| Surface half span .....            | Bhalf = 16.446 ft             |
| Surface span .....                 | B = 32.893 ft                 |
| Surface half area .....            | Shalf = 92.28 ft <sup>2</sup> |
| Surface total area .....           | Stot = 184.56 ft <sup>2</sup> |
| Surface LE sweep angle .....       | GLE = 2.805°                  |
| Surface aspect ratio .....         | AR = 5.8621                   |
| Surface taper ratio .....          | TR = 0.5586                   |
| Surface Mean Aerodynamic Chord ... | Cmac = 5.761 ft               |
| Surface MAC location .....         | Xmac = 0.365 ft               |
| .....                              | Ymac = 7.447 ft               |



This information can be used when calculating the CG and neutral point locations as percentages of the Mean Aerodynamic Chord (MAC). For instance, the CG located at  $X_{cg} = 2.0465$  ft becomes  $100 \cdot (2.0465 - 0.365) / 5.7 = 29.5\%$  MAC.

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**Figure 8-4. Computing the neutral point.**

The neutral point was estimated by computing the slope of the CMY curve for two different values of Xcg; 1.85 ft and 2.0465 ft. The corresponding values of CMY for two angles-of-attack (AOA1 and AOA2) was evaluated (**SURFACES** provides a tool to make this simple, shown in Figure 8-4). The resulting Xneu is 2.721 ft. This corresponds to:

$$\frac{X_{NEU}}{C_{REF}} = 100 \frac{(2.721 - 0.365)}{5.7} = 41.3\%MAC$$

### V8.3 Results from SURFACES

Summary (note that values from Nelson and Schmidt appear to be from the same source):

|                         |  | Ryan Navion       |                                                                         |                       |                                                   |                                                                                            |
|-------------------------|--|-------------------|-------------------------------------------------------------------------|-----------------------|---------------------------------------------------|--------------------------------------------------------------------------------------------|
|                         |  | Source/<br>Symbol | <i>Flight Stability<br/>and Automatic<br/>Control</i> , R. C.<br>Nelson | VLM using<br>SURFACES | Panel Method<br>using CMARC <sup>7</sup><br>(DWT) | <i>Introduction to<br/>Aircraft Flight<br/>Dynamics</i> , Louis<br>V. Schmidt <sup>8</sup> |
| Air density             |  | ρ                 | 0.002378 slugs/ft³                                                      |                       |                                                   |                                                                                            |
| Outside Air Temperature |  | OAT               | 518.69 °R                                                               |                       |                                                   |                                                                                            |
| Speed of sound          |  | ao                | 1116 ft/s                                                               |                       |                                                   |                                                                                            |
| Altitude                |  | H                 | 0 ft                                                                    |                       |                                                   |                                                                                            |
| Far field speed         |  | Vinf              | 178 ft/s                                                                |                       |                                                   |                                                                                            |
| Mach Number             |  | M                 | 0.159                                                                   |                       |                                                   |                                                                                            |
| Baseline AOA            |  | AOA               | -                                                                       | 0.88°                 | 0.76°                                             | 0.6°                                                                                       |
| Reference span          |  | Bref              | 33.40 ft                                                                |                       |                                                   |                                                                                            |

<sup>7</sup> Source: <http://www.aerologic.com/stab/corr.html>.

<sup>8</sup> Document is cited in footnote 1.

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|                                                 |       |                            |                                                  |           |        |
|-------------------------------------------------|-------|----------------------------|--------------------------------------------------|-----------|--------|
| Reference wing chord                            | Cref  | 5.70 ft                    |                                                  |           |        |
| Reference wing area                             | Sref  | 184.0 ft <sup>2</sup>      |                                                  |           |        |
| Reference aspect ratio                          | ARref | 6.06                       |                                                  |           |        |
| Reference weight                                | W     | 2750 lbs                   |                                                  |           |        |
| Center of gravity (along X-axis)                | Xcg   | 0.295 Cref                 | 0.295 Cref<br>(2.0465 ft)                        | 0.25 Cref | -      |
| Neutral point (along X-axis)                    | Xneu  | 0.552 Cref                 | 0.413 Cref<br>(2.721 ft)                         | 0.38 Cref | -      |
| Moment of inertia about X-axis                  | Ixx   | 1048 slugs-ft <sup>2</sup> |                                                  |           |        |
| Moment of inertia about Y-axis                  | Iyy   | 3000 slugs-ft <sup>2</sup> |                                                  |           |        |
| Moment of inertia about Z-axis                  | Izz   | 3530 slugs-ft <sup>2</sup> |                                                  |           |        |
| Product of inertia about X-axis                 | Ixy   |                            |                                                  |           |        |
| Product of inertia about Y-axis                 | Ixz   |                            |                                                  |           |        |
| Product of inertia about Z-axis                 | Iyz   |                            |                                                  |           |        |
| Lift coefficient for $\alpha = 0^\circ$         | CLo   | 0.375 <sup>9</sup>         | 0.329                                            | -         | -      |
| Slope of lift coefficient                       | CLA   | 4.44                       | 4.722                                            | 5.15      | 4.44   |
| Lift coefficient                                | CL    | 0.41                       | 0.406                                            | 0.415     | 0.415  |
| Parasitic drag coefficient                      | CDo   | 0.0390                     | 0.039                                            | -         | -      |
| Total drag coefficient                          | CD    | 0.05                       | 0.04952 (from<br>quadratic drag<br>polar)        | -         | 0.051  |
| Slope of drag coefficient at $\alpha = 0^\circ$ | CDA   | 0.33                       | 0.258 (quantic fit<br>at $\alpha = 0.88^\circ$ ) | -         | 0.330  |
| Span efficiency (Oswald's)                      | e     | 0.85                       | 0.85                                             | -         | -      |
| AOA DERIVATIVES                                 | CXA   | 0.330                      | 0.268                                            | -         | -      |
|                                                 | CYA   | -                          | 0                                                | -         | -      |
|                                                 | CZA   | -4.850                     | -4.935                                           | -         | -      |
|                                                 | CMXA  | -                          | 0                                                | -         | -      |
|                                                 | CMA   | -0.683                     | -0.584                                           | -0.8721   | -0.683 |
|                                                 | CMZA  | -                          | 0                                                | -         | -      |
| AOY DERIVATIVES                                 | CXB   | -                          | 0                                                | -         | -      |
|                                                 | CYB   | -0.564                     | -0.5065                                          | -         | -      |
|                                                 | CZB   | -                          | 0                                                | -         | -      |
|                                                 | CLB   | -0.0740                    | -0.07723                                         | -         | -      |
|                                                 | CMB   | -                          | 0                                                | -         | -      |
|                                                 | CNB   | 0.0710                     | 0.07639                                          | -         | -      |
| d(AOA)/dt DERIVATIVES                           | CXTA  | -                          | Not predicted                                    | -         | -      |
|                                                 | CYTA  | -                          | Not predicted                                    | -         | -      |
|                                                 | CZTA  | 1.7000                     | Not predicted                                    | -         | -      |
|                                                 | CLTA  | -                          | Not predicted                                    | -         | -      |
|                                                 | CMTA  | -4.36                      | Not predicted                                    | -         | -      |
|                                                 | CNTA  | -                          | Not predicted                                    | -         | -      |
| U-DERIVATIVES                                   | CXU   | -0.1000 (?)                | Not predicted                                    | -         | -      |
|                                                 | CYU   | -                          | Not predicted                                    | -         | -      |
|                                                 | CZU   | -                          | Not predicted                                    | -         | -      |
|                                                 | CLU   | -                          | Not predicted                                    | -         | -      |
|                                                 | CMU   | -                          | Not predicted                                    | -         | -      |
|                                                 | CNU   | -                          | Not predicted                                    | -         | -      |
| P-DERIVATIVES                                   | CXP   | -                          | 0                                                | -         | -      |
|                                                 | CYP   | -                          | 0                                                | -         | -      |
|                                                 | CZP   | -                          | 0                                                | -         | -      |
|                                                 | CLP   | -0.41                      | -0.44                                            | -         | -      |
|                                                 | CMP   | -                          | 0                                                | -         | -      |
|                                                 | CNP   | -0.0575                    | -0.0652                                          | -         | -      |
| Q-DERIVATIVES                                   | CXQ   | -                          | 0                                                | -         | -      |
|                                                 | CYQ   | -                          | 0                                                | -         | -      |
|                                                 | CZQ   | -                          | -8.99                                            | -         | -      |
|                                                 | CLQ   | -                          | 0                                                | -         | -      |
|                                                 | CMQ   | -9.96                      | -12.98                                           | -6.87     | -9.96  |

<sup>9</sup> From analysis on page 54 of reference document.

|               |     |         |         |   |   |
|---------------|-----|---------|---------|---|---|
| R-DERIVATIVES | CLQ | -       | 0       | - | - |
|               | CXR | -       | 0       | - | - |
|               | CYR | -       | 0.4059  | - | - |
|               | CZR | -       | 0       | - | - |
|               | CLR | 0.1070  | 0.1374  | - | - |
|               | CMR | -       | 0       | - | - |
|               | CNR | -0.1250 | -0.1557 | - | - |

Additional comparison based on a table from the source <http://www.aerologic.com/stab/corr.html>.

|                                    | CMARC<br>(DWT) | Perkins &<br>Hage | Etkin  | Seckel <sup>10</sup> | Datcom | SURFACES | Wind<br>Tunnel | Flight Test |
|------------------------------------|----------------|-------------------|--------|----------------------|--------|----------|----------------|-------------|
| Angle of<br>attack, $\alpha$       | -0.732         | -                 | -      | -                    | -      | -0.732°  | -              | -           |
| Elevator<br>deflection, $\delta_e$ | 8.28           | -                 | -      | -                    | -      | -        | -              | -           |
| $C_L$                              | 0.271          | -                 | -      | -                    | -      | 0.269    | -              | -           |
| $C_D$                              | -              | -                 | -      | -                    | -      | -        | -              | -           |
| $C_{L\alpha}$                      | 5.23           | 4.36              | 4.25   | 4.54                 | 5.5    | 4.722    | 4.52           | 6.04        |
| $C_{D\alpha}$                      | -              | -                 | -      | -                    | -      | -        | -              | -           |
| $C_{M\alpha}$                      | -0.91          | -0.83             | -0.715 | -0.545               | -1.24  | -0.584   | -0.95          | -           |
| $C_{Mq}$                           | -6.99          | -9.6              | -9.75  | -9.5                 | -12.98 | -12.98   | -              | -           |
| $C_{M\delta_e}$                    | -2.99          | -3.0              | -4.91  | -4.91                | -6.58  | -        | -              | -           |
| $d\epsilon/d\alpha$                | 0.428          | -                 | -      | -                    | -      | -        | -              | -           |
| $C_{L\delta_e}$                    | 0.66           | -                 | -      | -                    | -      | -        | -              | -           |
| $C_{M\delta_e}$                    | -1.68          | -                 | -      | -                    | -      | -        | -1.42          | -1.42       |

<sup>10</sup> Seckel E. "Stability and Control of Airplane and Helicopters", Academic Press, 1964.

## Validation 9 - Comparison to NACA R-1208

### V9.1 Introduction

This validation compares **SURFACES** analysis to the swept back wing featured in the NACA report R-1208. In the report a highly swept back, high aspect ratio wing compares three numerical methods to wind tunnel test results. In this validation sample, a similar approach will be taken and the section lift coefficients from **SURFACES** will be compared to the wind tunnel test results. The wing planform is shown in Figure 9-1.

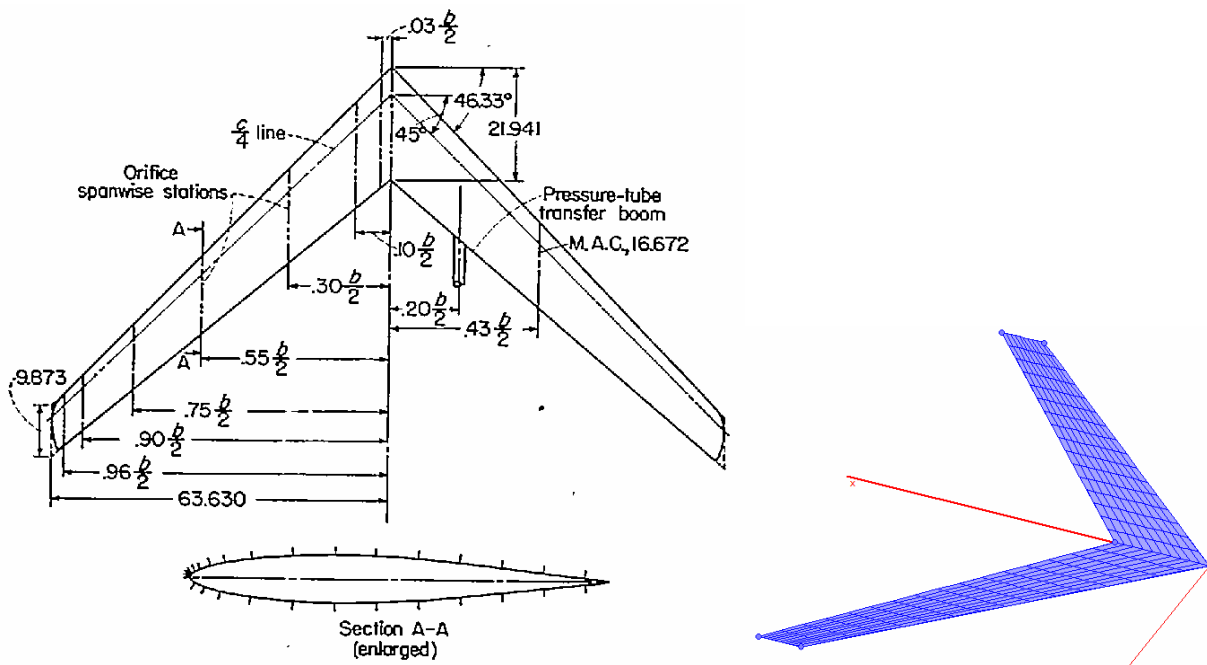


**Figure 9-1: The swept-back wing wind tunnel tested per NACA R-1208. Inserted image shows the SURFACES VL model.**

Three VL models were generated; one has 16 spanwise panel per wing side, the second one has 32 spanwise panels, and the third has 64 spanwise panels per side. The comparison takes place at  $4.7^\circ$  angle of attack, per the NACA report..

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### V9.2 Expected Result

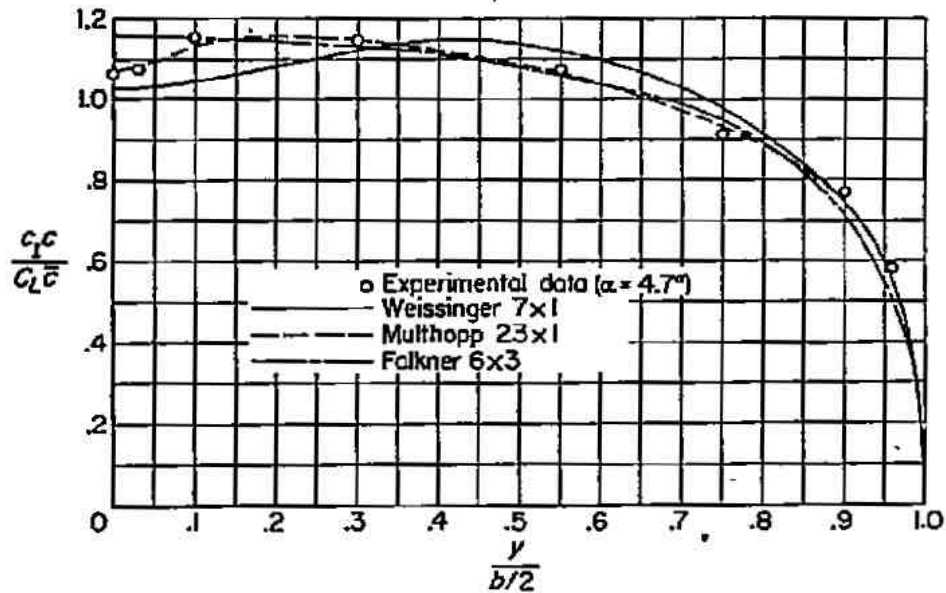


Figure 9-2: Original graph of spanwise loading from NACA R-1208.

### V9.3 Results from SURFACES

The comparison of the numerical to the experimental data shows a close agreement, but also that the accuracy improves with number of panels.

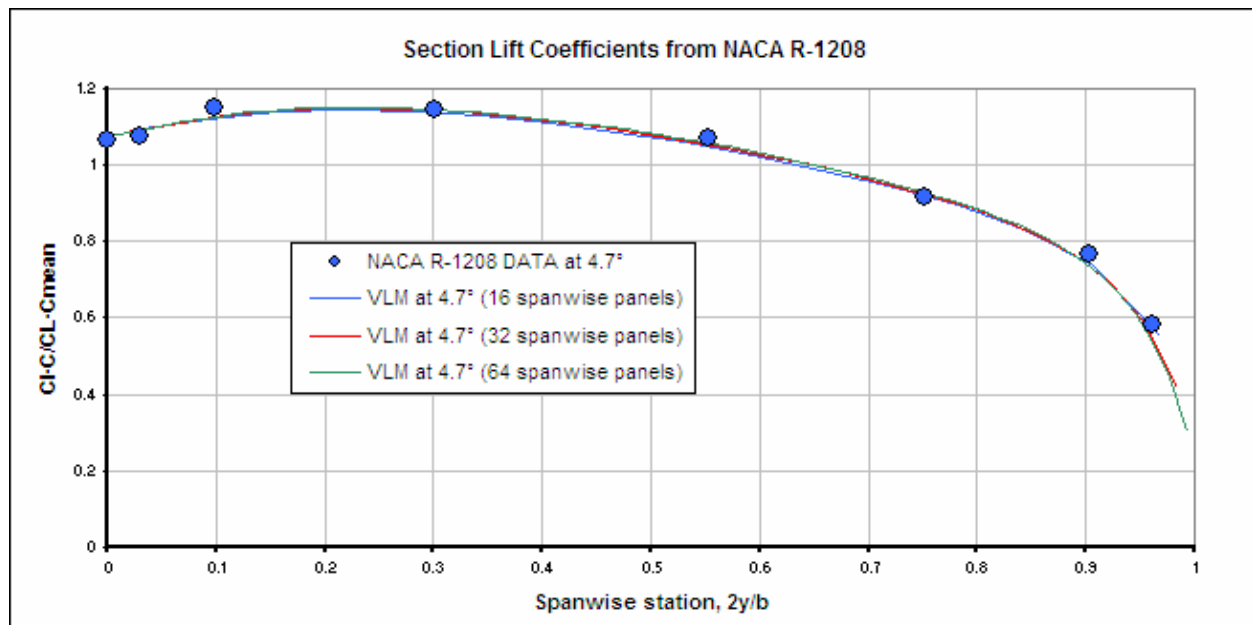


Figure 9-3: Comparing spanwise loading from **SURFACES** to experimental data from NACA R-1208.

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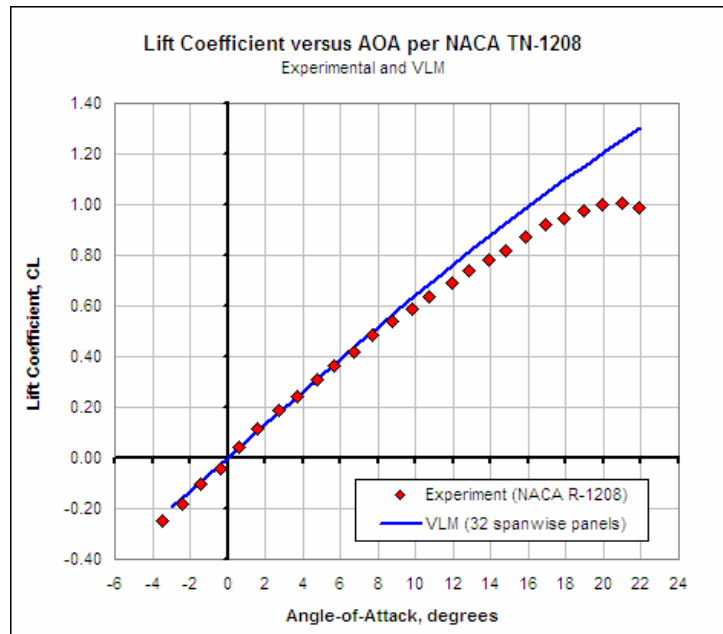


Figure 9-4: Comparing lift curve from **SURFACES** to experimental data from NACA R-1208.

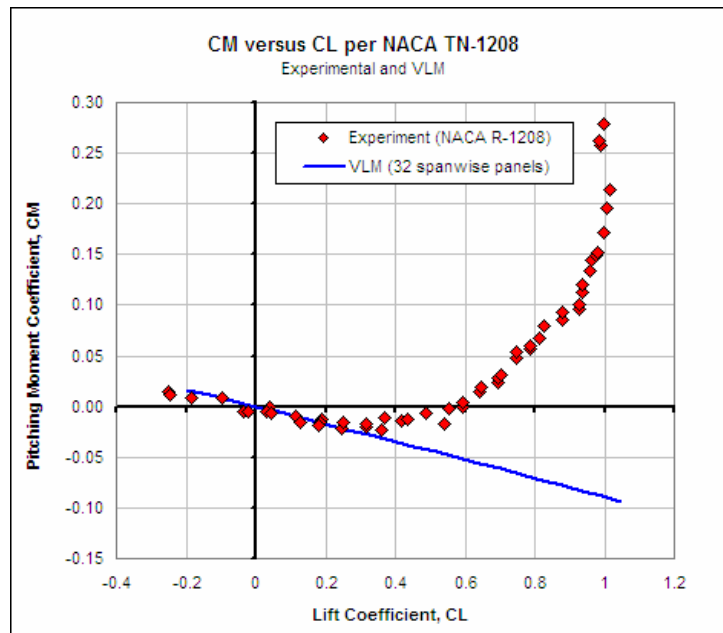


Figure 9-5: Comparing moment curve from **SURFACES** to experimental data from NACA R-1208.

The experimental data shows the well known early tip stall phenomena of swept back wings, caused by spanwise flow near the tips. This is reproduced here to remind the user that all inviscid codes (vortex-lattice, doublet-lattice, panel-codes, etc) do not model this viscous phenomena accurately because the mathematical solution forces the flow to stay attached.

## Validation 10 - Comparison to NACA TN-1422

### V10.1 Introduction

This validation compares **SURFACES** analysis to two of the three tapered and twisted wings featured in the NACA report TN-1422. This report compares several aerodynamic properties of three wings obtained in wind tunnel tests. In this validation sample the section lift coefficients, lift curves, and moment curves for two of these wings (from hereon referred to as WING 2 and WING 3) from **SURFACES** will be compared to the wind tunnel test results. The general planform shape is shown in Figure 10-1, and is reproduced from the original document.

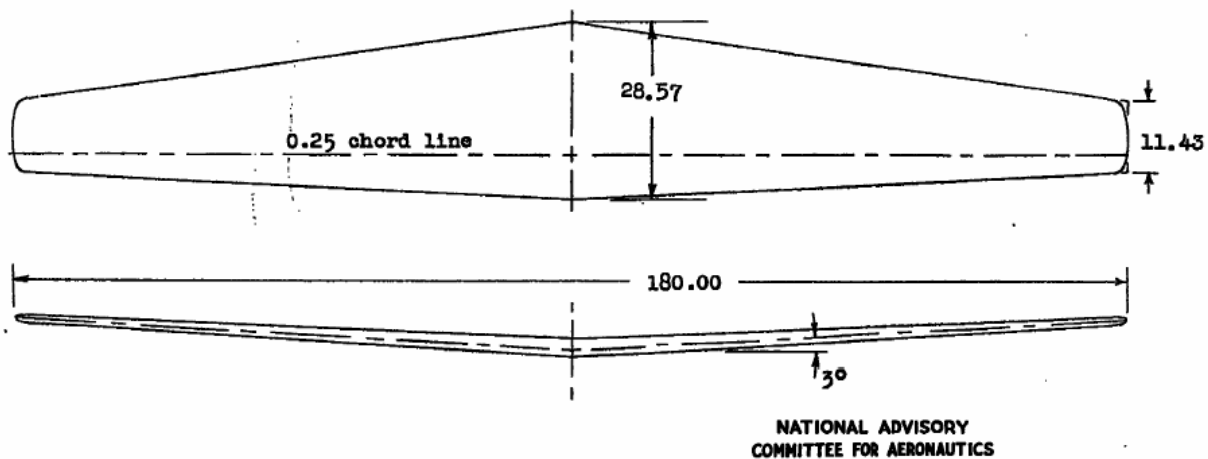


Figure 1.- General dimensions of 10-percent-thick wings tested in Langley 19-foot pressure tunnel. Aspect ratio, 9; ratio of root chord to tip chord, 2.5. (All dimensions are in inches.)

Figure 10-1: The general shape of the wind tunnel model tested per NACA TN-1422.

### V10.2 Results from **SURFACES**

The comparison of the numerical to the experimental data shows a close agreement.

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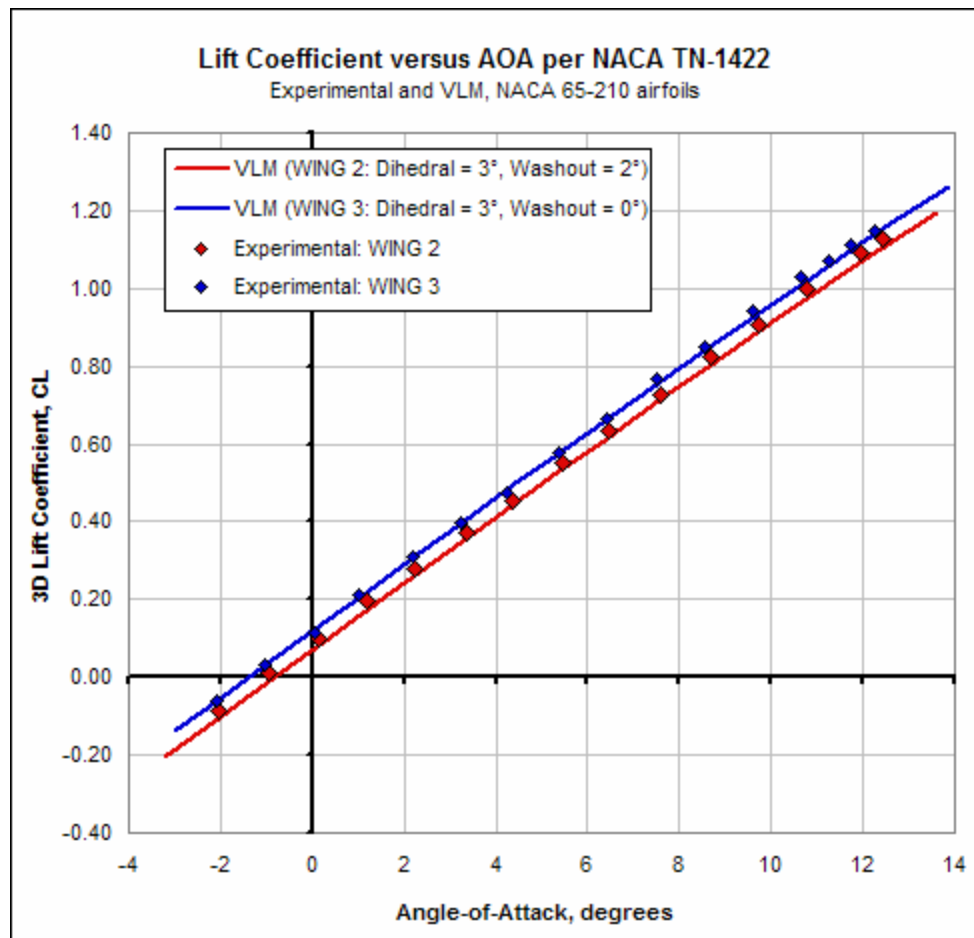


Figure 10-2: Match for the lift curve for the twisted and untwisted wings.

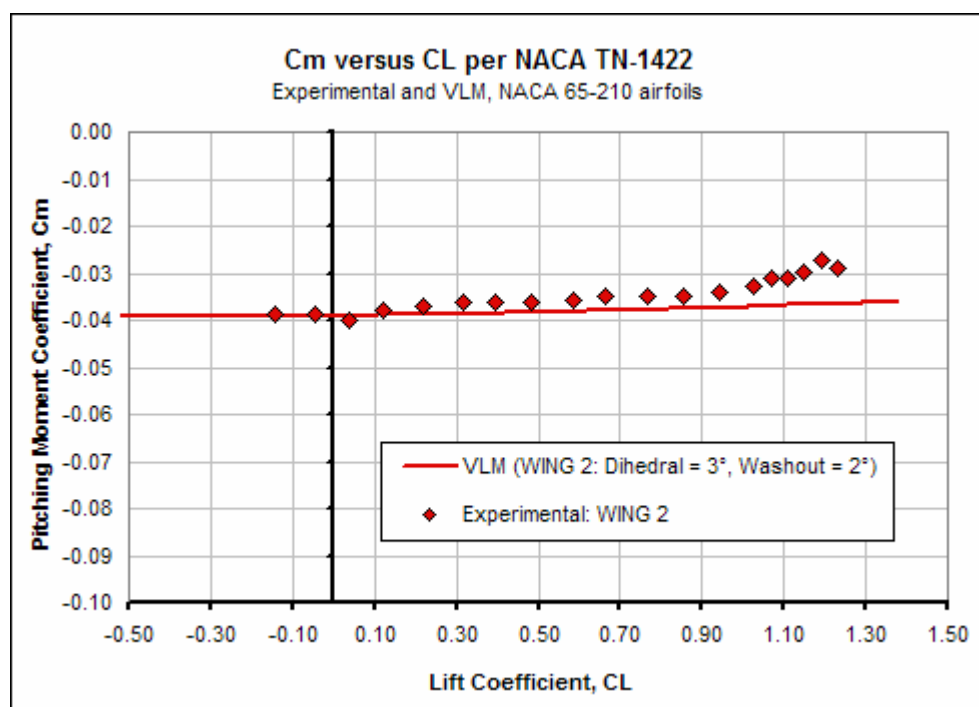


Figure 10-3: Match for the pitching moment for the twisted wing. Note the deviation at higher values of the lift coefficient, which is caused by viscous effects.

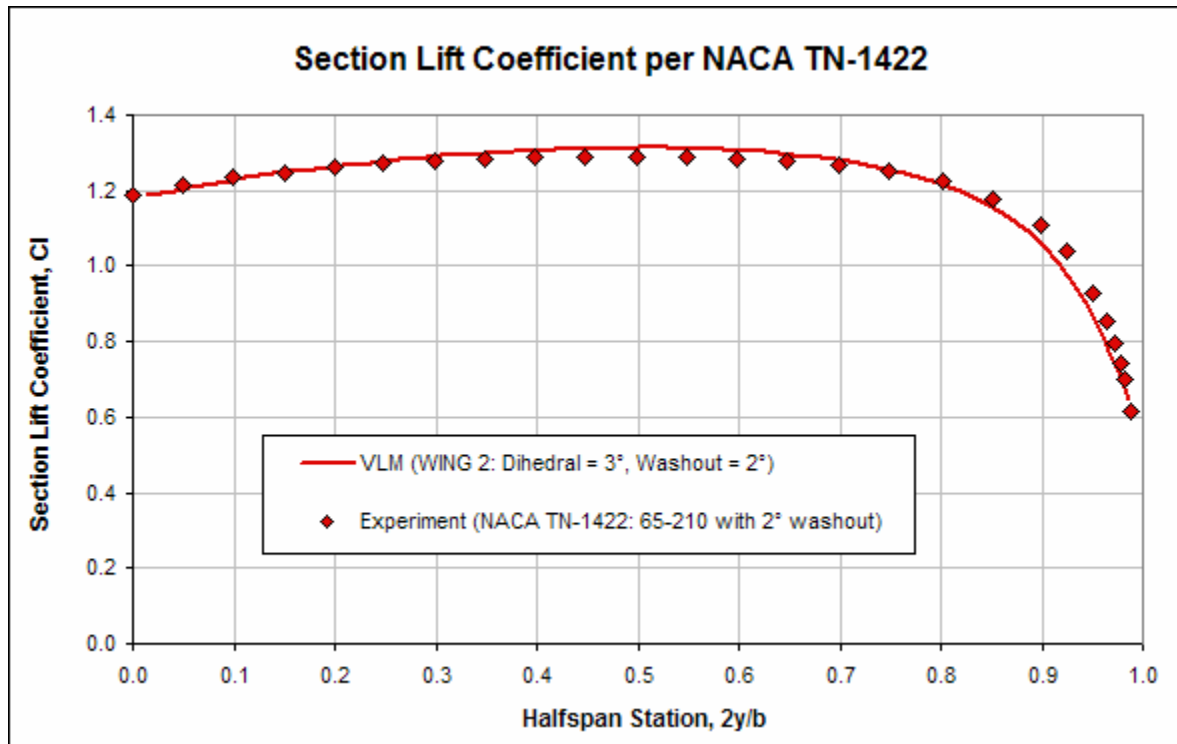


Figure 10-4: Lift distribution at stall for the twisted wing.

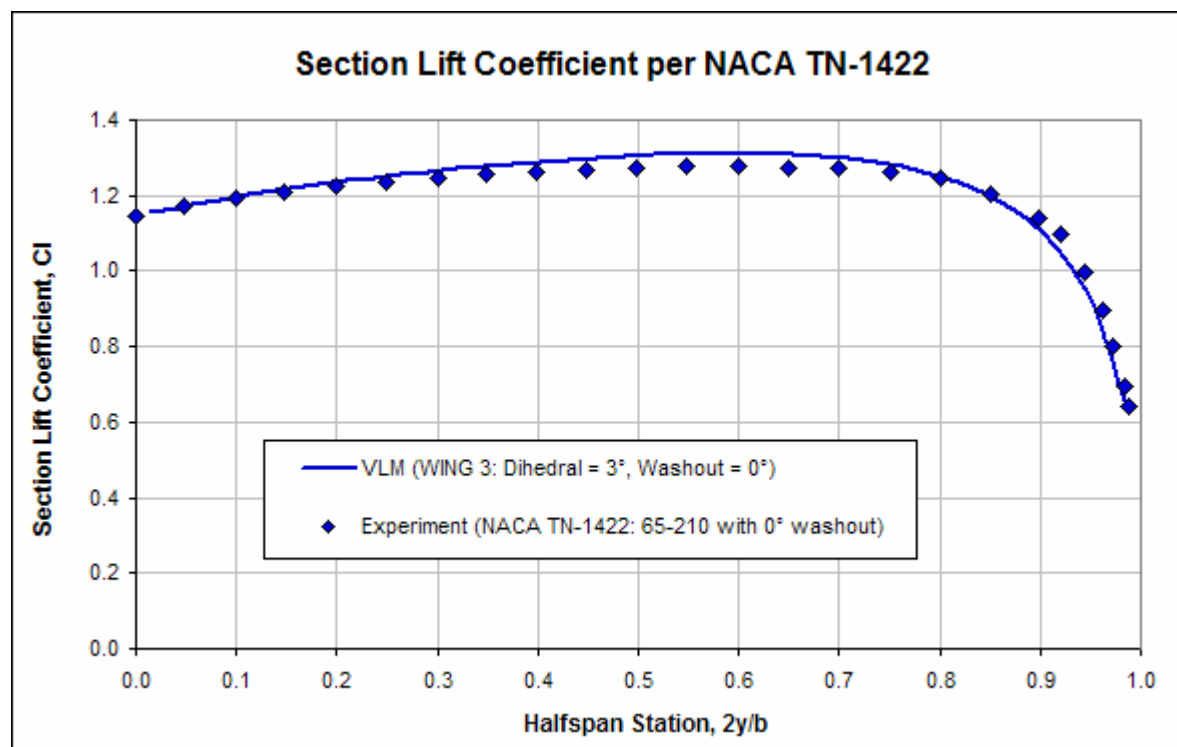


Figure 10-5: Lift distribution at stall for the untwisted wing.